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PHASE II FINAL REPORT

VOLUME IB - PART II
STUDY RESULTS

SYSTEM TECHNOLOGY ANALYSIS OF AEROASSISTED ORBITAL TRANSFER VEHICLES: MODERATE LIFT/DRAG (0.75-1.5)

AUGUST 1985

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GENERAL ELECTRIC

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FORWARD

This final report of the "System Technology Analysis of Aeroassisted Orbital Transfer Vehicles: Moderate Lift/Drag (0.75-1.5)" was prepared by the General Electric Company, Space Systems Division for the National Aeronautics and Space Administration's George C. Marshall Space Flight Center (MSFC) in accordance with Contract NAS8-35096. The General Electric Company, Space Systems Division was supported by the Grumman Aerospace Corporation as a subcontractor during the conduct of this study. This study was conducted under the direction of the NASA Study Manager, Mr. Robert E. Austin, during the period from October 1982 through June 1985.

The first phase of this program focused on a ground based AOTV and was completed in September 1983. The second phase was directed towards a space based AOTV and the cryofueled propulsion subsystem-configuration interactions and was completed in March of 1985. The second phase was jointly sponsored by NASA-MSFC and the NASA Lewis Research Center (LeRC). Dr. Larry Cooper was the LeRC study manager.

This final report is organized into the following three documents:

Volume IA	Executive Summary - Parts I & II
Volume IB	Study Results - Parts I & II
Volume II	Supporting Research and Technology Report
Volume III	Cost and Work Breakdown Structure/ Dictionary

Part I of these volumes covers Phase 1 results, while Part II covers Phase 2 results.

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VOLUME IB - PART II

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STUDY RESULTS - PART II

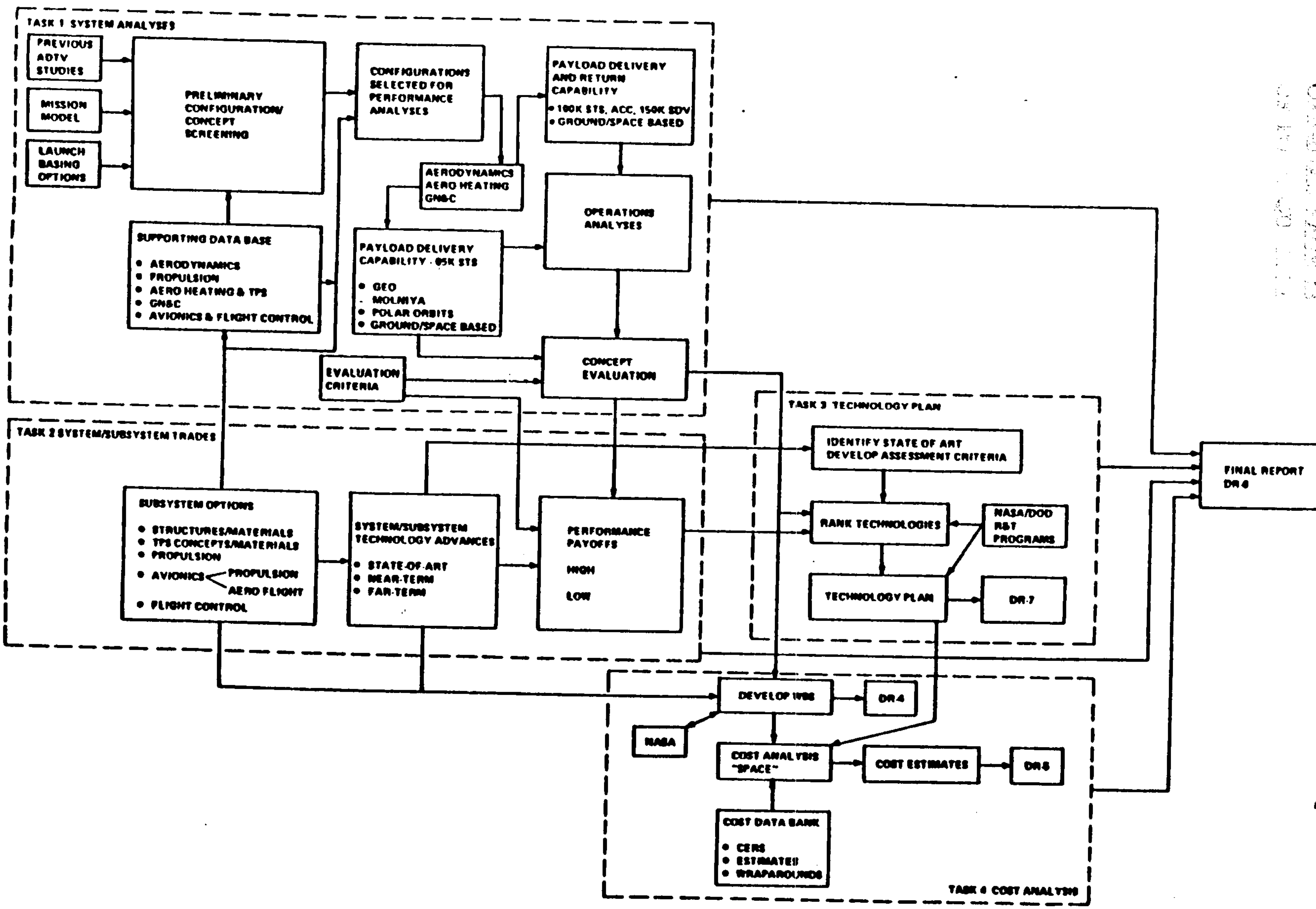
1.0 INTRODUCTION

Technology payoffs of representative ground based Mid L/D AOTVs have been assessed and prioritized in Phase I of this study. These results have been summarized in Part I of this final report. Phase II of this study was directed towards identification and priorization of technology payoffs of representative space based Mid L/D AOTVs and the cryofueled propulsion subsystem - configuration interactions.

Part II of this volume contains a complete compilation of the results from Phase II of this study. The compilation format for this volume is charts/figures and facing page text (Kent format).

The major tasks for this portion of the study are outlined in the Figure . Task structure for this Phase of the study closely parallels that of Phase I, with two exceptions. Phase I emphasized ground based AOTVs, whereas Phase II emphasizes space based AOTVs. In addition, approximately 50% of the Phase II effort was focused on LOX-H_2 fueled propulsion subsystems-configuration interactions. This expanded propulsion subsystem task had been advocated by the NASA Lewis Research Center.

MID L/D Φ II STUDY FLOW DIAGRAM



The primary issues confronting the Space Based AOTV are broken out into Systems and Technology areas and are listed in the figure.

Space basing of an AOTV opens up numerous configuration opportunities which were explored in this study. AOTV size can exceed the launch vehicle cargo bay envelope by resorting to assembly in orbit. AOTV stage dry weight or gross lift off weight can exceed the Earth-LEO launch vehicle capability. With the absence of fully fueled tanks, as in the ground based configuration, much lighter gossamer type structures are possible on a Space Based AOTV that may result in performance gains. At the space station, payload rearranging or manifesting may prove attractive.

SPACE BASED MID L/D AOTV ISSUES

SYSTEMS ISSUES

- o WHAT SIZE AOTV
  - SHAPE/SIZE
  - PAYLOAD DELIVERY CAPACITY
  - UNIVERSAL AOTV VS. UNIQUE DELIVERY AND MANNED VEHICLE VS. ADAPTABLE VEHICLES
- o OPERATIONAL MODE
  - SINGLE STAGE (OPTIONAL USE OF L/D FOR PLANE CHANGE FLYING BELOW OVERSHOOT BOUND)
  - PERIGEE KICK + APOGEE KICK PROPULSION (CAN ALWAYS FLY NEAR OVERSHOOT BOUND)

TECHNOLOGY ISSUES

- o PAYLOAD MANIFESTING
  - AOTV OPERATIONAL MODE
  - AOTV BASING MODE
  - AOTV PREFERRED PROPELLANT SYSTEM
- o SPACE STATION UTILITY
  - PAYLOAD MANIFESTING
  - PROPELLANT STORAGE AND RESUPPLY
- o AOTV MAIN PROPELLANT SELECTION
  - LOX-H_2
  - N_2O_4 - MMH
  - OTHER
- o AERODYNAMICS
  - REDUCING W/C_LA, TO FLY HIGHER (AND COOLER?)
- o AEROTHERMODYNAMICS
  - TRANSITION TO TURBULENT FLOW (FLY HIGHER TO STAY LAMINAR?)
  - TOTALLY NON-CATALYTIC COATING
  - BASE HEATING TO ENGINE NOZZLES
- o PROPULSION SUBSYSTEM
  - NUMBER OF ENGINES
  - ENGINE THRUST
  - NOZZLE STRENGTH
  - NOZZLE PROTECTION
  - I_{SP}

2.0 SUMMARY

2.1 Summary of Phase I Ground Based AOTV Results

The major study objectives of Phase I of this program are summarized.

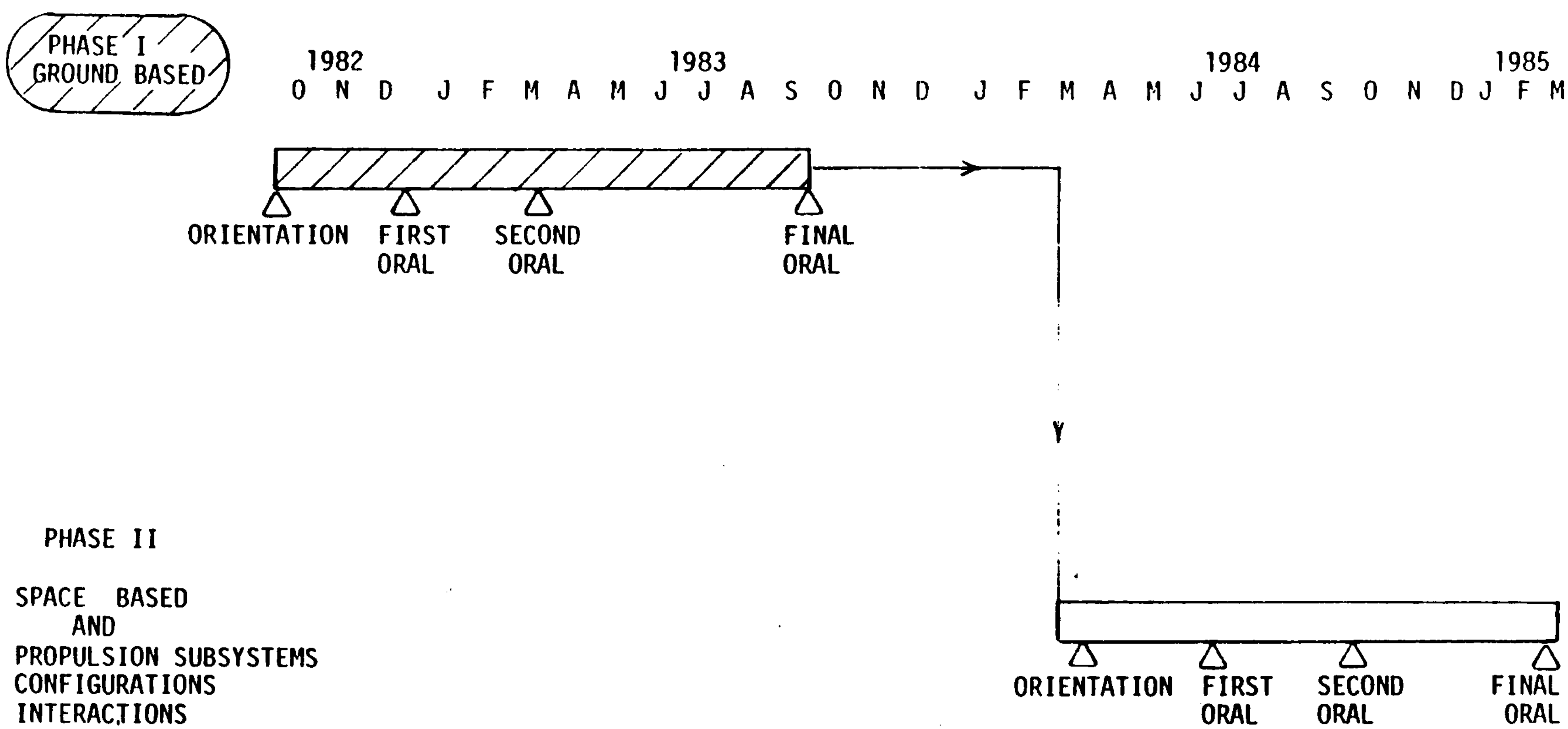
MID L/D AOTV SYSTEM TECHNOLOGY MAJOR STUDY OBJECTIVES

- PERFORM BROAD CONCEPT EVALUATIONS
- SYSTEMATICALLY ASSESS THE TECHNOLOGY REQUIREMENTS AND MISSION PARAMETERS FOR AEROASSISTED OTV'S OVER A RANGE OF HYPERSONIC L/D FROM 0.75 TO 1.5
- IDENTIFY TECHNOLOGY STATE-OF-ART
- IDENTIFY POTENTIAL TECHNOLOGY ADVANCES
- EVALUATE PERFORMANCE PAYOFFS FROM TECHNOLOGY ADVANCES
- GENERATE TECHNOLOGY PLAN AND PRIORITIZE

The first phase of this program focused on a ground based AOTV and was completed in September 1983. The second phase program was directed toward a space based AOTV and the cryofueled propulsion subsystem-configuration interactions and was completed in March of 1985. Major milestones are illustrated.

RE-ENTRY SYSTEMS OPERATIONS

MID L/D AOTV SYSTEM TECHNOLOGY SCHEDULE



Many combinations of basing and launch vehicle options, staging scenarios, missions (delivery, servicing, or manned round trip), and target orbits have been considered. The areas of primary emphasis in Phase I of this study are identified in the Figure. Major conclusions of Phase I will be summarized.

AOTV PERMUTATIONS AND COMBINATIONS

PHASE I EMPHASIS

AEROASSIST OPTIONS	BASING OPTIONS	LAUNCH VEHICLE OPTIONS	STAGING SCENARIOS	MISSION	TARGET ORBITS
ALL PROPULSIVE	GROUND	STD STS	SINGLE	DELIVERY	GEO
AEROBRAKED	SPACE	100K STS	1 1/2	SERVICING	6 HR POLAR
AEROMANEUVER		ACC	2	MANNED ROUND TRIP	MOLYNIA
L/D < 0.75		SDV			5X GEO
0.75 < L/D < 1.5					
L/D >1.5					

PHASE II EMPHASIS

PLUS EVALUATE PROPULSIVE SUBSYSTEM - CONFIGURATION INTERACTIONS

Configuration trade studies were conducted for the deliver and manned round trip missions. Various staging scenarios were evaluated, advantages of technology advances explored, and utility of hypersonic L/D determined. In order of importance, the most significant item was utilization of the AOTV as a perigee kick stage with the orbit inject propulsion built as an integral part of the payload. The next most important item was incorporation of advanced technology into the AOTV. Utilization of the hypersonic L/D for large plane change missions, a higher mixture ratio to reduce the size of the hydrogen tank and a shallow aft frustum angle to provide greater packaging efficiency completes the list of items having major impact.

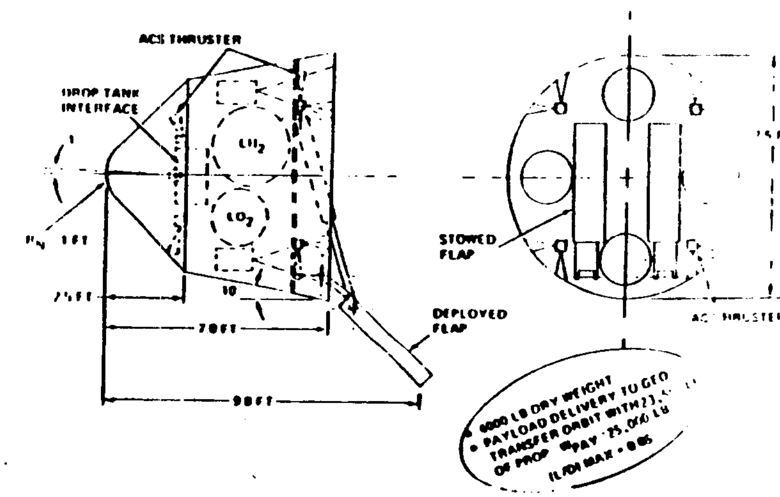
RE-ENTRY SYSTEMS OPERATIONS

PHASE I MID L/D CONFIGURATION TRADE CONCLUSIONS

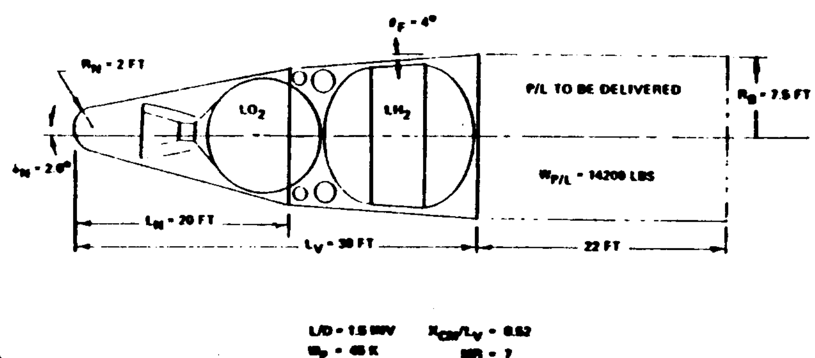
AOTV PERFORMANCE DRIVEN BY (IN ORDER OF IMPORTANCE)

- STAGING/DROP TANK SCENARIO - DELIVERY MISSIONS
- ADVANCED TECHNOLOGY
- L/D FOR LARGE PLANE CHANGE MISSIONS
- PROPULSION MIXTURE RATIO & AFT FRUSTUM ANGLE

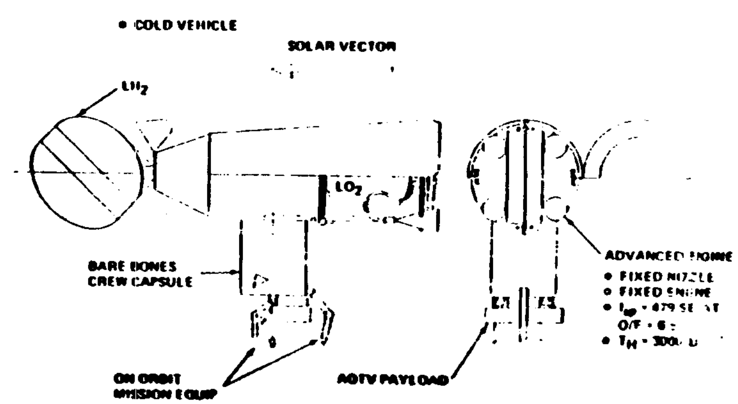
MINIMUM LENGTH AOTV "OH-3'



A 38 FT GEO DELIVERY VEHICLE



SMALL MANNED AOTV "H1": ORBITAL OPERATIONS



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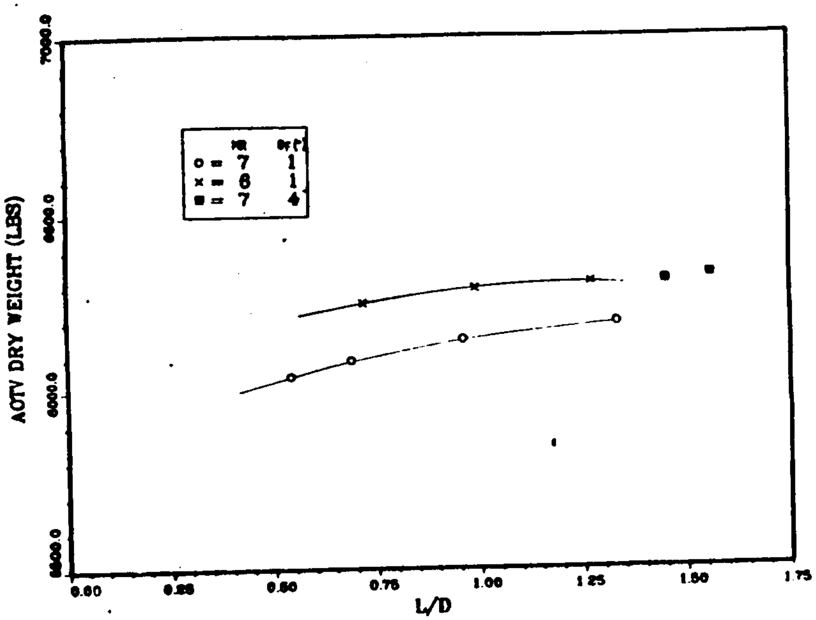
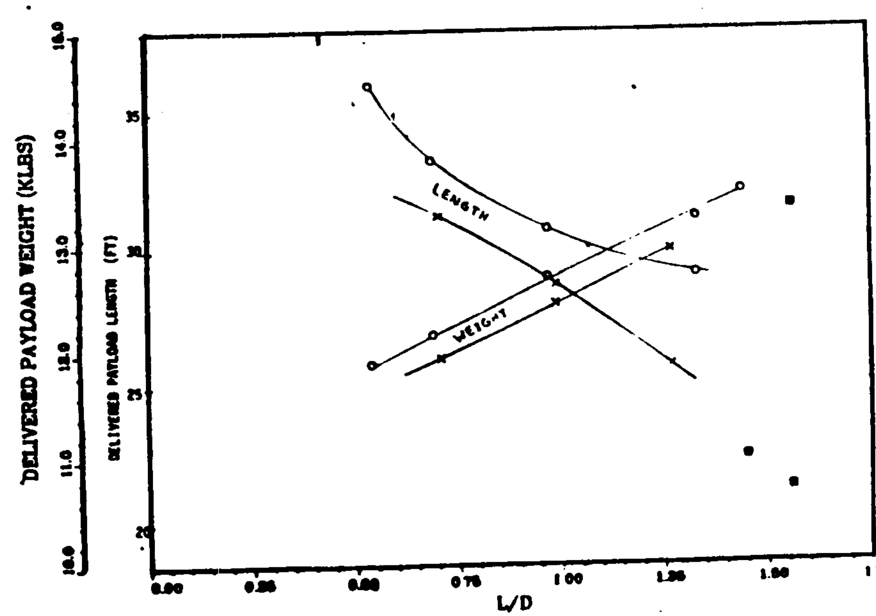
Some interesting ground based configuration trends are illustrated. The length constraints of the Shuttle payload bay and the lifting capacity to orbit create the trends illustrated of AOTV delivered payload length and weight to a six hour polar orbit. The trend with L/D is much greater for this polar orbit than for a geosynchronous orbit.

General Electric

Re-entry Systems Operations

DELIVERED PAYLOAD WEIGHT & LENGTH FOR INTERNAL TANKED AOTV – SIX HOUR POLAR





The advanced technology areas identified in Phase I of this study which offer significant benefit to ground based AOTV are listed.

GROUND BASED MID L/D AOTV ADVANCED TECHNOLOGY RECOMMENDATIONS

o SMALL NEW CRYOFUELED ADVANCED ENGINE

-- HIGH CHAMBER PRESSURE
-- ISP $\longrightarrow$ 480 TO 490 SEC
-- MIXTURE RATIO 6 TO 7

o EXTERNAL THERMAL PROTECTION SUB-SYSTEM IMPROVEMENTS

-- REDUCED COATING WEIGHT
-- INCREASED BOND/STRUCTURE AND MAXIMUM SURFACE OPERATING TEMPERATURES
-- NON CATALYTIC COATINGS

o AERODYNAMIC & AEROTHERMODYNAMIC KNOWLEDGE

-- CFD DEVELOPMENT
-- GROUND AND FLIGHT TEST CONFIRMATION/CALIBRATION

o STRUCTURE SUB-SYSTEM IMPROVEMENTS

-- IMPROVED DESIGN ALLOWABLES
-- ADVANCED MATERIALS

o AVIONICS & ELECTRICAL POWER SUBSYSTEM WEIGHT REDUCTION

-- DEGREE OF AUTONONY/REDUNDANCY
-- NEW MATERIALS

A detailed review of the current state-of-the-art in the various technology and subsystems areas was conducted to serve as a baseline point of departure for this study. Technology advancement possibilities identified in numerous recent studies of OTV, AOTV, SDV, and STS were reviewed. These results are compared with our in-house data base and parameters selected that represent improvements due to nominal expected growth resulting from normal funding of these technology areas. A number of these improvements resulting in from 10 to 70% reduction of subsystem weight are summarized in the figure. Other improvements include increase of maximum operating temperature of the thermal protection system elements, and others that will be identified in the discussion on subsystem trades.

TECHNOLOGY ADVANCEMENT POTENTIAL

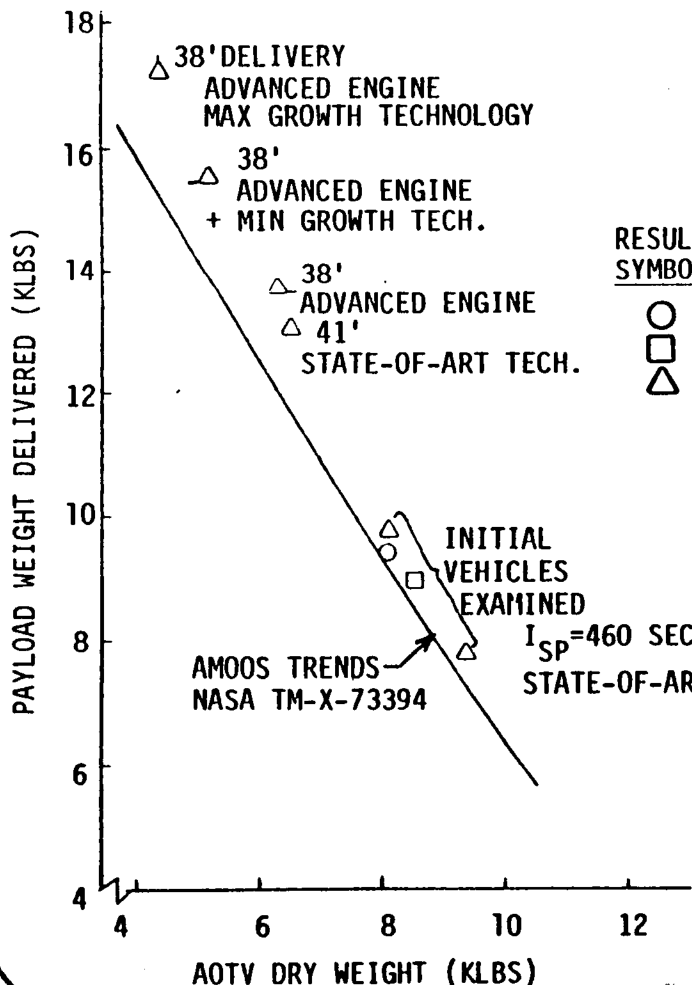
AOTV TECHNOLOGY ELEMENT	EXPECTED IMPROVEMENT
STRUCTURE SUBSYSTEM (SHELL, FRAMES, SUPPORTS AND FLAPS) - IMPROVED DESIGN ALLOWABLES - ADVANCED MATERIALS	10 TO 30% WEIGHT REDUCTION
THERMAL PROTECTION SUBSYSTEM - REDUCED COATING WEIGHT - INCREASED BOND/STRUCTURE TEMPERATURE - INCREASED MAX SURFACE TEMPERATURE CAPABILITY - NON CATALYTIC COATING	UP TO 56% WEIGHT REDUCTION
TRANSPIRATION COOLED NOSE - CAN LAMINAR FLOW BE MAINTAINED?	7^{o} PLANE CHANGE INCREASE FOR 5X GEO RETURN
AVIONICS SUBSYSTEM - DEGREE OF AUTONOMY/REDUNDANCY - ADVANCED COMPONENTS	50 TO 70% WEIGHT REDUCTION
ELECTRICAL POWER SUPPLY SUBSYSTEM - INCORPORATE NON METALLICS IN FUEL CELL - BACKUP BATTERY IMPROVEMENTS	20 TO 38% WEIGHT REDUCTION
NEW CRYOFUELED ENGINE - SMALL NOZZLE - HIGH CHAMBER PRESSURE	I_{SP} UP TO 480 SECONDS
- MIXTURE RATIO	6 TO 7
AERODYNAMICS - IMPROVED KNOWLEDGE ~ CFD	PROVIDE LESS CONTINGENCY IN STEERING ALGORITHMS FOR ERRORS IN AERO CHARACTERISTICS
AEROTHERMODYNAMICS - IMPROVED KNOWLEDGE ~ CFD	MINIMIZE SAFETY MARGIN ON TPS TYPICALLY 25%

AOTV dry weight also has a substantial effect on the payload delivery capability. The combined effects of AOTV hypersonic L/D and dry weight are summarized in the figures for the polar and GEO delivery missions for state-of-the-art and growth technology vehicles.

RE-ENTRY SYSTEMS OPERATIONS

MID L/D AOTV SINGLE STAGE DELIVERY CAPABILITIES

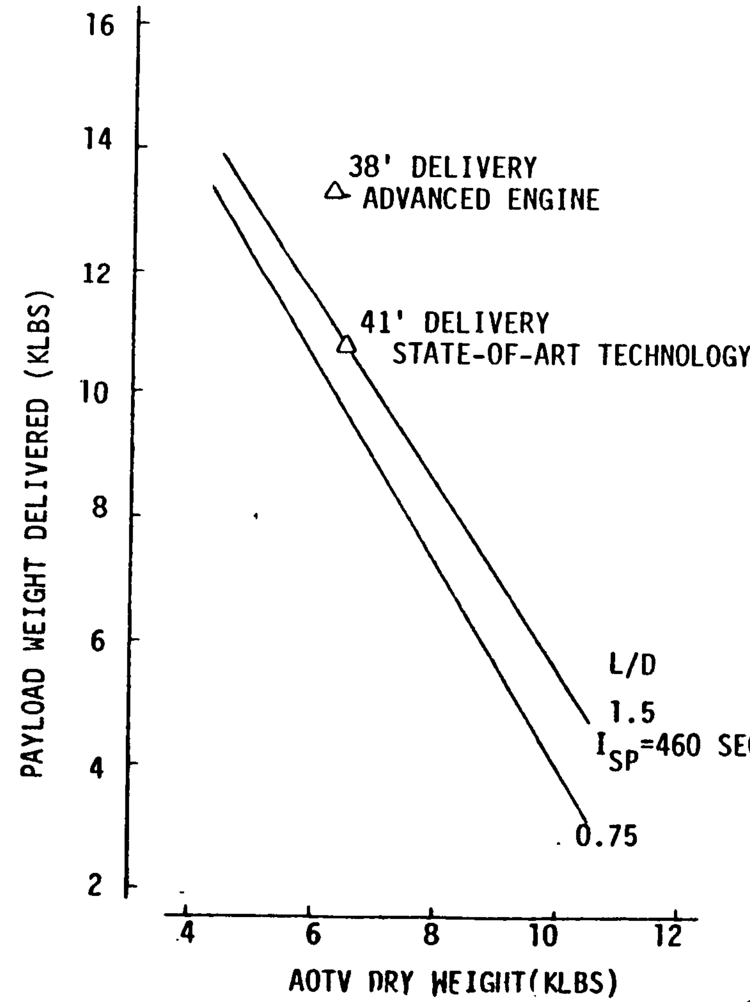
SINGLE STAGE GROUND BASED VEHICLES



RESULTS OF THIS STUDY

SYMBOL	L/D	AOTV GLOW
○	0.75	62K
□	1.04	62K
△	1.5	62K

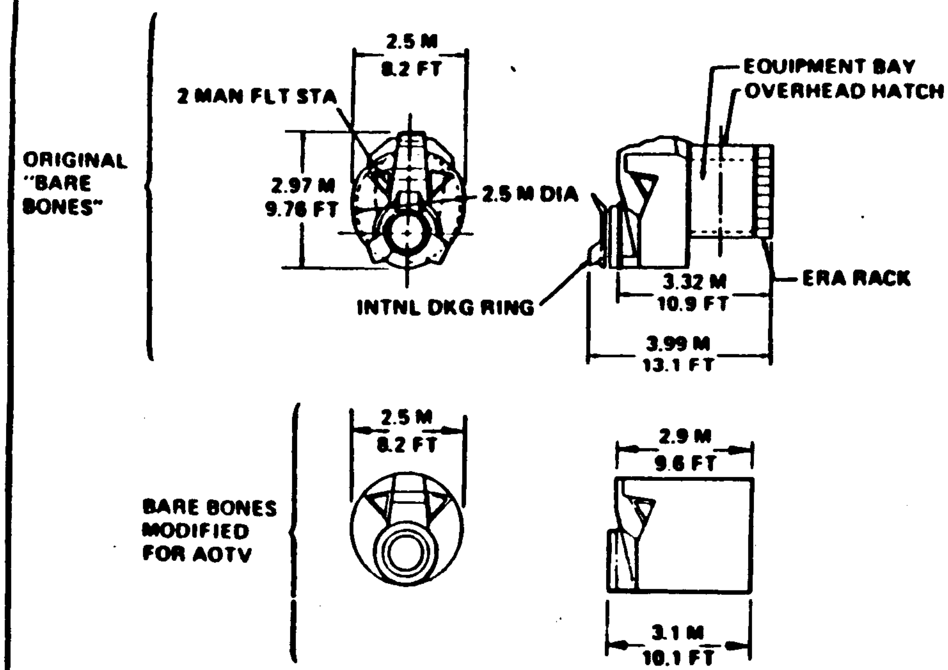
6-HR POLAR DELIVERY CAPABILITY
SINGLE STAGE GROUND BASED VEHICLES
$i=59^{o}$



Two crew capsule designs were reviewed for possible use in this AOTV study; 1) the final baseline "functional minimum" Manned Orbital Transfer Vehicle (MOTV) capsule and 2) a "bare bones" derivative of the Lunar Module (LM) ascent stage capsule. The two capsules were both designed during the MOTV study. The "functional minimum" capsule was designed to sustain a crew of 2 during missions which lasted as long as 30 days. The "bare bones" was designed to sustain a crew of 2 for 6 day missions. Two thirds of the MOTV missions require 6 days (or less) to complete a mission. Several of the >6 day mission contain much time for changing locations at GEO, but less than 6 days for working. The "bare bones" capsule is judged adequate for a significant number of projected GEO missions. A modification which preserved the same volume and weight as the LM derivative shown in the figure was selected for use in this study due to its smaller size and weight. The LM derivative weighs 3580 lbs, with on-board mission equipment, deployment and antirotation mechanisms bringing the total to 5300 lbs.

RE-ENTRY SYSTEMS OPERATIONS

LUNAR MODULE DERIVATIVE BARE BONES CABIN



- 3580 LB DRY WEIGHT
  - WEIGHT DERIVED FROM HARDWARE, LUNAR MODULE ASCENT STAGE #12
- BARE BONES CABIN ADEQUATE FOR CURRENTLY ENVISIONED NASA MISSIONS TO GEO
  - EXCLUDES LONG DURATION CONSTRUCTION MISSIONS
- BARE BONES DESIGNED FOR SIX DAY MISSIONS
  - 30 DAY MISSIONS (WITH ONLY A FEW WORKING DAYS E.G., MULTIPLE SATELLITE SERVICING) ARE PRACTICAL
  - CREW EFFICIENCY IMPAIRMENT IS EXPECTED ON LONG MISSIONS

Most past OTV studies did not select a totally new cryogenic engine for their vehicles. Rather, they selected a modification to the RL-10 which would raise the engine's specific impulse from 447 seconds to 460 seconds.

The acceptance of a minimum man rating criteria like "fail safe" precludes the use of a single engine for manned vehicles. Some members of the engineering community do not want to apply man rating criteria to the main propulsion system. The logic presented in the figure is addressed to them. and attempts to use statistical arguments to show that a single engine is not economically viable for manned vehicles.

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PROPULSION IMPLICATIONS OF MAN-RATING: HISTORY

STATISTICAL APPROACH

- GAC, NOV 1979, "MANNED ORBITAL TRANSFER VEHICLE", VOL 3
  - RISK TO MOTV CREWMEMBER FOR 10 FLIGHTS SET EQUAL TO CAREER RISK OF COMMERCIAL AIRLINE PILOT
  - MOTV CREW RISK 1/50 PER FLIGHT → 1 CATASTROPHE IN 1000 OTV MISSIONS
- BOEING, 1980, "ORBIT TRANSFER VEHICLE CONCEPT DEFINITION STUDY", VOL 3
  - SINGLE ENGINE DOES NOT MEET RELIABILITY REQUIREMENTS OF 1/50 RISK
- P&W, 1980 "ORBIT TRANSFER VEHICLE ENGINE STUDY, TASKS 8-12 REPORT"
  - FUNDING LIMITS WILL PREVENT DEMONSTRATION OF REQUIRED RELIABILITY (0.99959)

CONCLUSIONS:

- MULTIPLE ENGINES REQUIRED FOR MAN-RATED AOTV
- RL 10 DERIVATIVE II TOO LARGE FOR MULTI-ENGINE USE ON EARLY AOTV
- PRELIMINARY ENGINE TYPE SELECTION
  - ADVANCED EXPANDER CYCLE, ISP ~ 480 SEC

At conferences and meetings which occurred during the Phase I AOTV studies, significant differences of opinion arose among members of the engineering community on the subject of man rating an OTV and the minimum number of engines required for a man rated OTV. The position of this study team on some of these issues is summarized in the figure.

SUMMARY OF POSITION ON MAN RATED AOTV

- CREW AT GEO MUST BE RETURNED TO LEO
- SINGLE ENGINE AT GEO IS UNACCEPTABLE
  - NO WINGS & NO PARACHUTES AT GEO
- CREW PROTECTION OUTSIDE AOTV PROGRAM IS NOT RELIABLE
  - DEDICATED RESCUE VEHICLE NOT CREDIBLE ALL THE TIME
  - TEMPORARY GEO HOUSING ("LINE SHACK") TOO EXPENSIVE
- AOTV TECHNOLOGY STUDY MAN RATING CRITERIA:
  - "AOTV WILL SURVIVE ANY TWO INDEPENDENT NONEXPLOSIVE FAILURES & RETURN CREW TO ORBITER"
    - PRESSURE VESSELS, PRIMARY STRUCTURE, & TPS ARE EXEMPT

FAIL SAFE/FAIL SAFE

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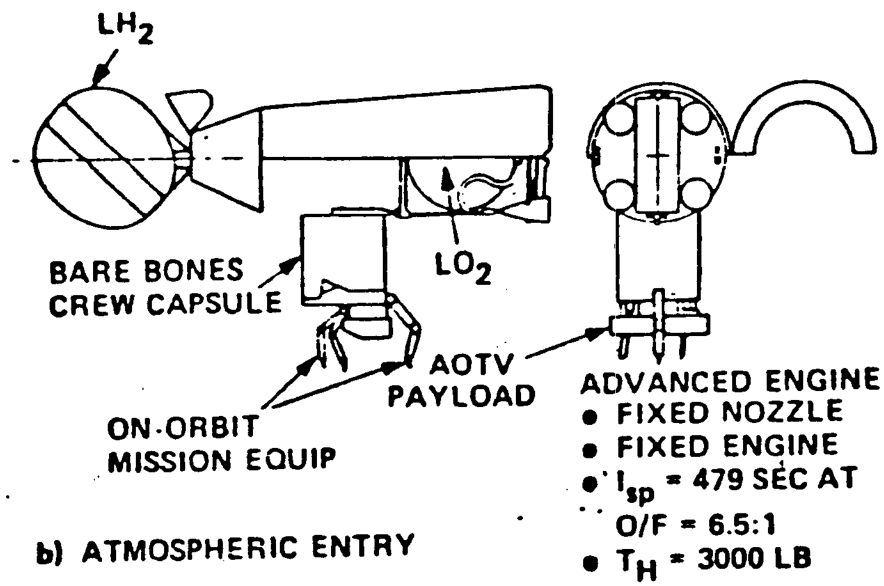
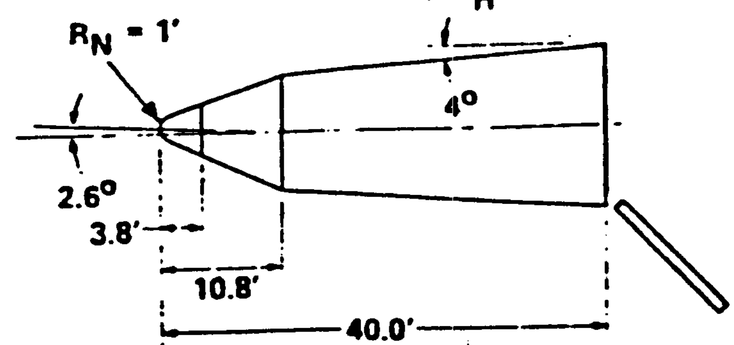
A manned mission to GEO on one shuttle launch is made possible by use of the H-1M configuration. The "H-1M" mission is enabled by two advanced technologies. Aerodynamic braking on return to LEO saves the propellant necessary to decelerate the AOTV by about 8000 ft/sec. This manned mission to GEO would be impossible (on one shuttle launch) if that propellant had to be carried on the mission. The L/D of 1.04, which permits 14.6^{o} of aerodynamic change of orbital plane, allows a 2100 pound payload to be delivered. If all plane changing (28.5^{o}) had to be performed propulsively, the additional fuel would have eliminated the entire 2100 pound payload. Similarly, advanced propulsion technology also enables this mission. The specific impulse of current RL10 engines is 447 seconds. To deliver 2100 pounds of payload to GEO on the "H-1M" mission, 479 seconds of I_{sp} is required in addition to the L/D = 1.04. To perform the "H-1M" mission with no payload to GEO, 479 seconds of specific impulse is required. More than high performance is necessary for "H-1M" engines. For a given thrust level and nozzle expansion ratio, a large number of small engines provides a significantly smaller AOTV than would a single large engine. The smaller AOTV weighs less and has a better chance of fitting within a single 60-foot long orbiter cargo bay. Also, multiple engines can provide a level of redundancy that is highly desirable (fail safe) for manned missions.

Small Manned AOTV "H-1M"

a) ORBITAL OPERATIONS

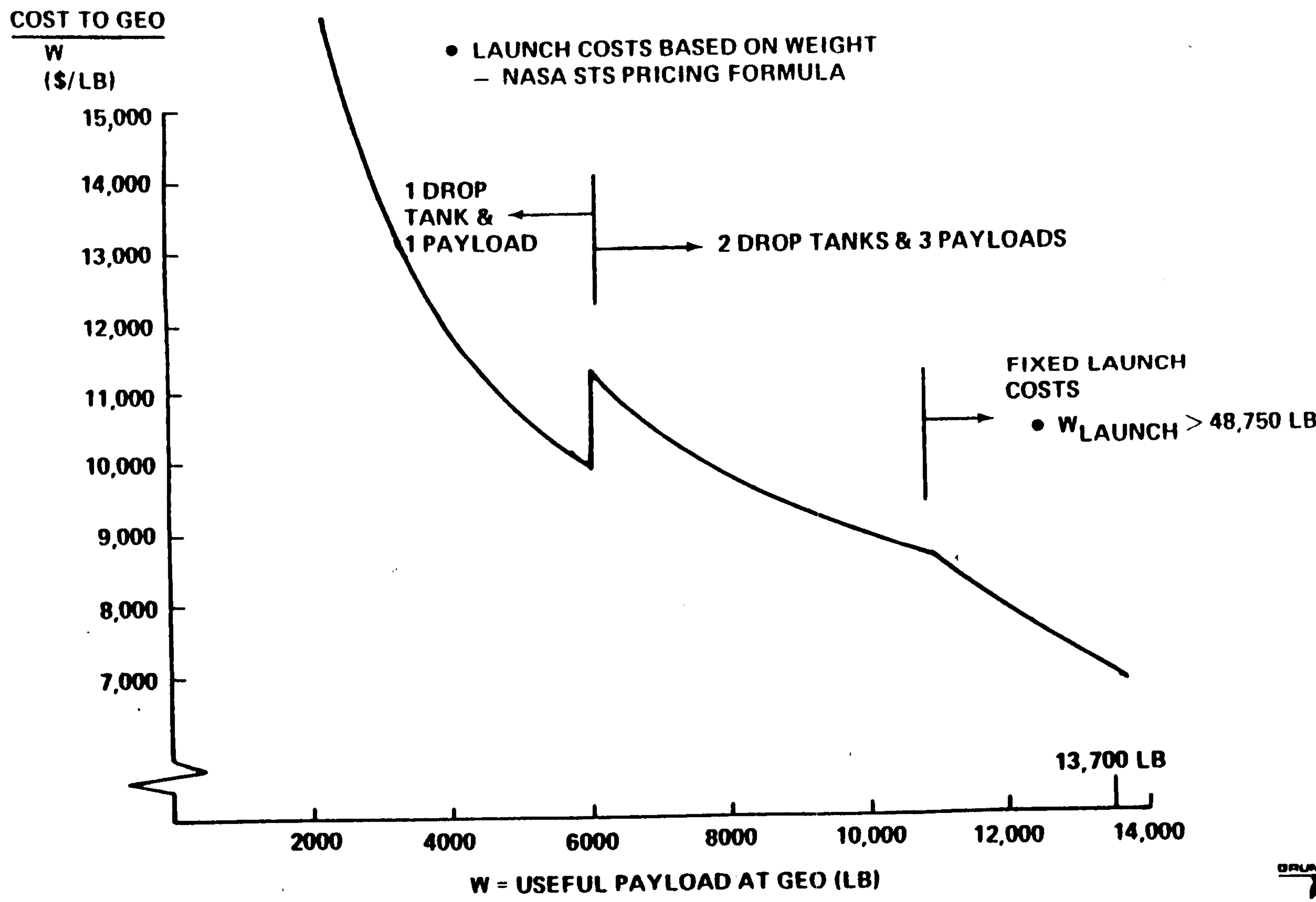


b) ATMOSPHERIC ENTRY



A Work Breakdown Structure (WBS) and related project costs have been developed for selected AOTV options. Phases covered include DDT&E, production, and operations. Program cost estimates include only those incurred by the prime contractor. Operation costs and earth to LEO transport charges have been converted into payload delivery costs for GEO delivery. A summary of these cost results is presented in the figure for the OH-3.

USER COSTS VS PAYLOAD WEIGHT: GROUND BASED OH-3



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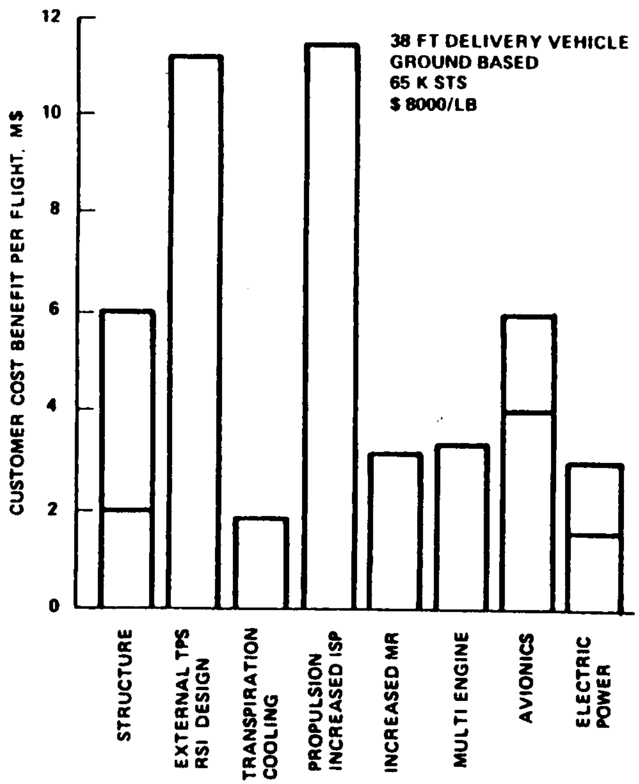
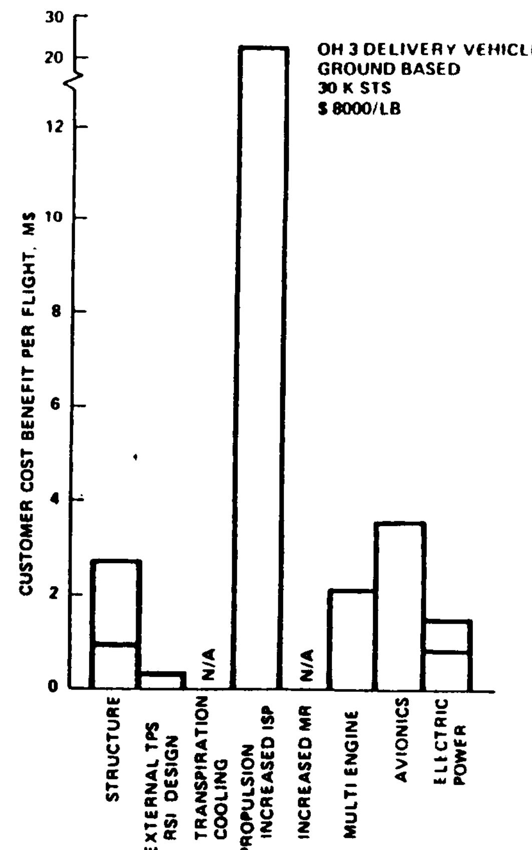
Various techniques exist for ranking the technology benefits. The method selected for this study is as follows: given a subsystem weight reduction or other performance improvement possibility, the effect on increased payload weight was determined and this payload gain was converted to a customer cost benefit (i.e., money saved by the paying customer on Shuttle launch charges) assuming a nominal delivery cost to GEO of $8000 per lb. The mid L/D AOTV payload delivery sensitivities have been combined with delivery cost and subsystem weight reduction possibilities to generate the results summarized in the figure for the 38 ft and OH-3 delivery vehicles. Note that the 38 ft single stage vehicle has very different technology payoffs from the small OH-3 staged vehicle. However, both vehicles benefit substantially from high I_{sp} engines ($\sim$480 sec).

RE-ENTRY SYSTEMS OPERATIONS

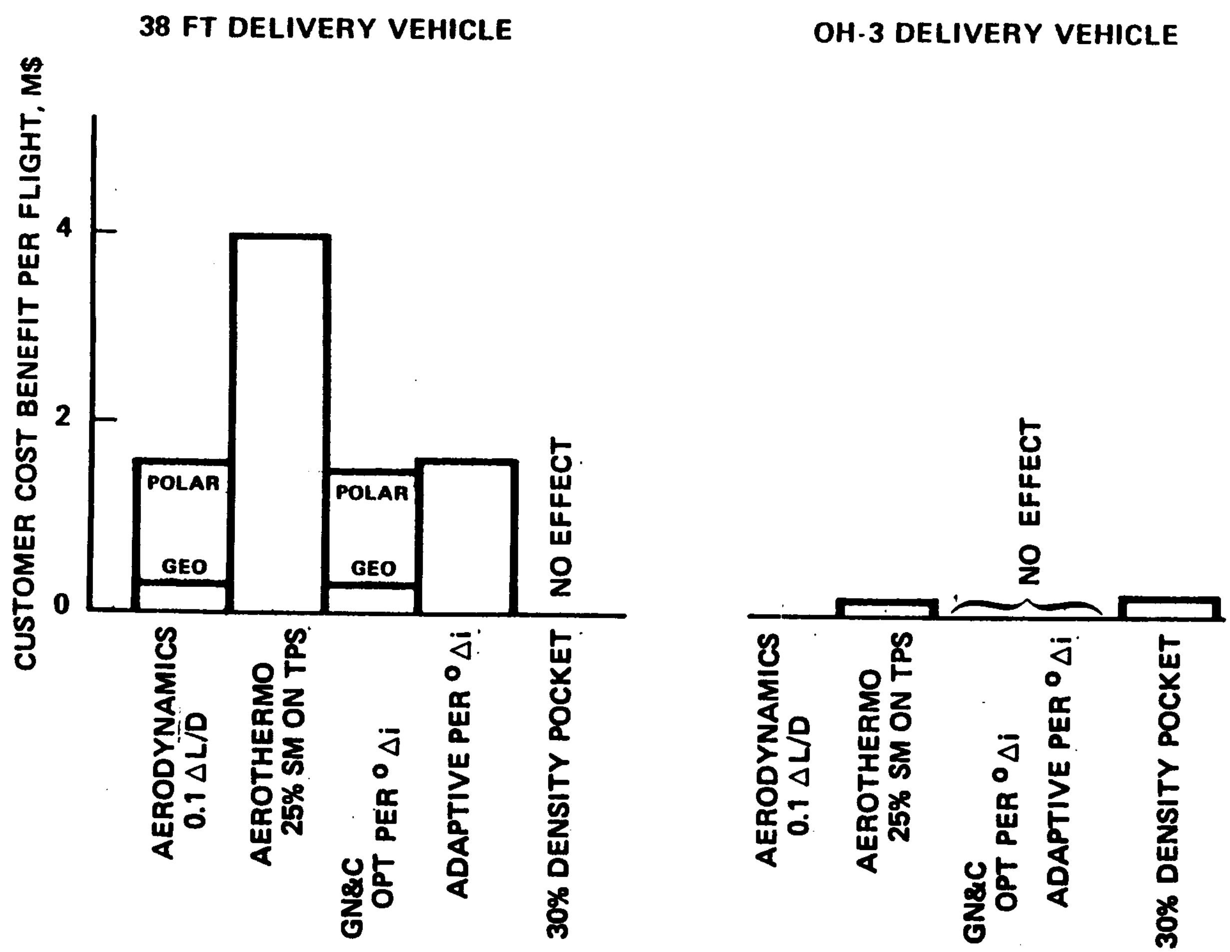
EFFECT OF SUBSYSTEM WEIGHT REDUCTION DUE TO TECHNOLOGY ADVANCES ON CUSTOMER COST BENEFIT





Additional technology advance benefits are summarized in the Figure for both vehicles. Aerodynamic uncertainties due to viscous and rarefaction effects will exist and could amount to as much as ± 0.1 of Δ L/D. This uncertainty requires a propellant contingency which in turn decreases the payload delivery capability. Flight vehicles have typically flown initially with a safety margin in the thermal protection system of as much as 25%. This translates into a very large payload loss (and hence cost benefit if it is decreased or eliminated) for the 38 ft delivery vehicle but a much smaller effect for the OH-3 vehicle due to its much smaller size. In the GN&C subsystem areas, the ability to obtain aerodynamic plane change is translated into payload gain and hence customer cost benefit. The value of an "optimum" guidance system that has been selected because it is capable of obtaining the most aerodynamic plane change from a given vehicle configuration is illustrated for one degree of incremental plane change. The value of an "Adaptive" guidance system that has the capability of updating during the early portion of entry is illustrated for each additional one degree of incremental plane change. The value of an "Adaptive" guidance system that has the capability of updating during the early portion of entry is illustrated for each additional one degree of plane change that can be generated. The effect of encountering a 30% density shear (pocket) similar to that experienced by a recent STS flight has been demonstrated to have no effect on a vehicle with L/D = 1.5 but to have a small effect on a vehicle with L/D = 0.6.

EFFECT OF TECHNOLOGY ADVANCES ON CUSTOMER COST BENEFIT



Technology payoff benefits quantified in Phase I of this program all fall in the mission enhancing category. These benefits have been rank ordered in decreasing order of importance. Some of the technology advance benefits such as aerodynamic and aerothermodynamic knowledge and GN&C techniques were more nebulous to quantify but were included in the prioritized list at a rank deemed appropriate.

GROUND BASED
MID L/D AOTV TECHNOLOGY PRIORITIES

MISSION ENABLING TECHNOLOGY	
NONE	
MISSION ENHANCING TECHNOLOGY	
PRIORITY	ITEM
1*	SMALL NEW ENGINE WITH INCREASED I_{SP} & MR
2	EXTERNAL TPS DESIGN
3	AVIONICS WEIGHT REDUCTION
4	AERODYNAMIC KNOWLEDGE
5	AEROTHERMODYNAMIC KNOWLEDGE
6	STRUCTURE WEIGHT REDUCTION
7	ELECTRICAL POWER SUPPLY WEIGHT REDUCTION

*PHASE II RESULTS INDICATE THAT STORABLE PROPELLANT ENGINES MAY BE MORE ATTRACTIVE

The major conclusions of the Ground Based mid L/D AOTV System Technology Analysis are listed.

MAJOR PHASE 1 GROUND BASED MID L/D STUDY CONCLUSIONS

- SUBSTANTIAL PERFORMANCE IMPROVEMENTS OBTAINABLE BY DEVELOPING "ENHANCING" TECHNOLOGIES
  - ADVANCED EXPANDER SMALL LOX-HYDROGEN ENGINE WITH SPECIFIC IMPULSE OF 480-490 SEC
  - EXTERNAL THERMAL PROTECTION SYSTEM WEIGHT REDUCTION
    - -- REDUCED COATING WEIGHT, INCREASED MAXIMUM ALLOWABLE BOND/STRUCTURE TEMPERATURE, DEVELOP NON-CATALYTIC COATING
  - STRUCTURAL SHELL WEIGHT REDUCTION
    - -- IMPROVED DESIGN ALLOWABLES AND ADVANCED STRUCTURAL MATERIALS
  - AVIONICS SUBSYSTEM WEIGHT REDUCTION
  - ELECTRIC POWER SUBSYSTEM WEIGHT REDUCTION
  - AEROTHERMODYNAMIC KNOWLEDGE
- SMALL VEHICLES USED IN PERIGEE KICK + AKP DELIVERY SYSTEMS HAVE VERY DIFFERENT TECHNOLOGY COST BENEFITS FROM LARGER SINGLE STAGE VEHICLES
- WITHIN ANY STAGING CLASS OF GROUND BASED VEHICLES, PERFORMANCE SENSITIVITIES AND TECHNOLOGY COST BENEFITS ARE INDEPENDENT OF L/D WITHIN THE MID L/D RANGE (.75 TO 1.5)
- USE OF MID L/D AOTV PROVIDES SIGNIFICANT AERODYNAMIC PLANE CHANGE CAPABILITY (20° FOR RETURN FROM GEO WITH L/D = 1.5) AND CONTROL AUTHORITY OVER TRAJECTORY DISPERSIONS AND OFF NOMINAL ATMOSPHERES
- ALL MID L/D AOTV "ENABLING" TECHNOLOGY IS READY TODAY.

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MAJOR PHASE 1 GROUND BASED MID L/D STUDY CONCLUSIONS

(CONTINUED)

- VERY SHORT UPPER STAGES (PERIGEE KICK + AKP) POSSIBLE WITH DROP TANKS
  - PERMIT DELIVERY OF 45'LONG PAYLOADS
  - 95% OF VEHICLE COST IS RETURNED FOR RE-USE
- PERIGEE KICK + AKP APPEARS MOST EFFICIENT FOR GEO DELIVERY
  - AVG $8000/LB FOR GEO DELIVERY SYSTEM OF MULTIPLE SATELLITES
- MANNED MISSION TO GEO POSSIBLE WITH ONE 65K STS LAUNCH
  - CREW OF TWO & TOOLS
  - ONE TON OF PAYLOAD REMAINS AT GEO
- MULTIPLE LOW THRUST FIXED ENGINES PROVIDE LOWEST WEIGHT FOR MANNED VEHICLES
- LIGHTWEIGHT ASE (~3000 LB) APPEARS FEASIBLE

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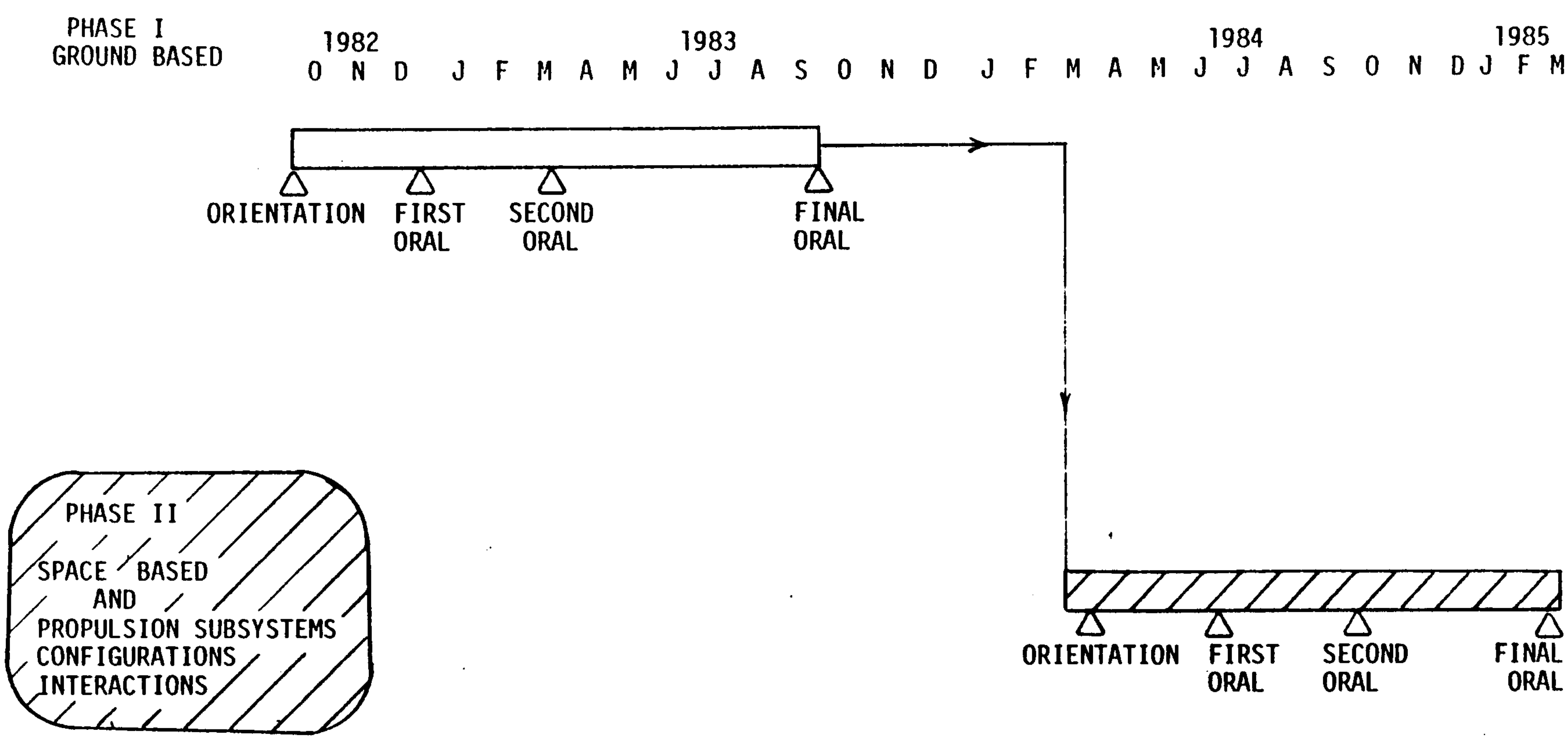
2.2 Summary of Phase II Space Based AOTV Results

The second phase of this program focused on a space based AOTV and the cryofueled propulsion subsystem-configuration interactions. This phase was completed in March 1985. Major milestones are illustrated.

RE-ENTRY SYSTEMS OPERATIONS

MID L/D AOTV SYSTEM TECHNOLOGY SCHEDULE



Many combinations of basing and launch vehicle options, staging scenarios, missions (delivery, servicing, or manned round trip) and target orbits have been considered. The areas or primary emphasis in Phase II of this study are identified in the figure.

AOTV PERMUTATIONS AND COMBINATIONS

PHASE I EMPHASIS

AEROASSIST OPTIONS	BASING OPTIONS	LAUNCH VEHICLE OPTIONS	STAGING SCENARIOS	MISSION	TARGET ORBITS
ALL PROPULSIVE	GROUND	STD STS	SINGLE	DELIVERY	GEO
AEROBRAKED	SPACE	100K STS	1 1/2	SERVICING	6 HR POLAR
AEROMANEUVER		ACC	2	MANNED ROUND TRIP	MOLYNIA
L/D < 0.75		SDV			5X GEO
0.75 < L/D < 1.5					
L/D > 1.5					

PHASE II EMPHASIS

PLUS EVALUATE PROPULSIVE SUBSYSTEM - CONFIGURATION INTERACTIONS

Space basing of an AOTV opens up numerous configuration opportunities. The size can now exceed the launch vehicle cargo bay envelope by resorting to assembly in orbit. The AOTV stage dry weight or gross lift off weight can exceed the launch vehicle capability. With the absence of fully fueled tanks during a ground based launch much lighter gossamer type structures are possible that may result in performance gains. At the space station, payload rearranging or manifesting may prove attractive.

CONFIGURATION OPPORTUNITIES UNIQUE TO SPACE BASING

- FIXED AOTV GEOMETRY CAN EXCEED LAUNCH VEHICLE ENVELOPE
- AOTV STAGE DRY WEIGHT AND/OR GLOW CAN EXCEED SHUTTLE LIFT CAPACITY
- GOSSAMER TYPE STRUCTURES FOR SPACE BASED LOADS
- PAYLOAD REARRANGING OR MANIFESTING

The major conclusions from Phase II of this study are enumerated.

MAJOR PHASE II SPACE BASED MID L/D STUDY CONCLUSIONS

SYSTEM ISSUES

- o GEO DELIVERY CAPABILITY OF STS TRANSPORTABLE MAXIMUM SIZE PERIGEE KICK + AKP IS FAR IN EXCESS OF CURRENT AOTV MISSION MODEL REQUIREMENTS
- o PERIGEE KICK AOTV + AKP PRODUCES MINIMUM RECURRING COSTS FOR GEO DELIVERY
- o PERIGEE KICK VEHICLES CAN FLY NEAR ONE PASS OVERSHOOT BOUND TO REDUCE PEAK SURFACE TEMPERATURES

AEROTHERMODYNAMICS

- o PEAK SURFACE TEMPERATURES OF MID L/D AOTV'S ARE SIGNIFICANTLY LOWER
  - NEAR OVERSHOOT BOUND ENTRY COMPARED TO LARGE PLANE CHANGE
  - TOTALLY NON-CATALYTIC SURFACE COATING COMPARED TO PARTIALLY NON-CATALYTIC OR FULLY CATALYTIC SURFACE
- o SUBSTANTIAL UNCERTAINTY EXISTS IN MAGNITUDE OF HYPERSONIC BASE HEAT TRANSFER AND HEAT TRANSFER TO PROTRUDING NOZZLES
  - CURRENT TECHNOLOGY SUGGESTS MINIMAL NOZZLE PROTRUSIONS INTO SEPARATED FLOW REGION
  - ADVANCED TECHNOLOGY MAY PROVIDE ENLARGED ALLOWABLE ZONE (CFD, GROUND TESTS, CALIBRATION OF METHODOLOGY)
  - FLAPS SHOULD BE MOVED ONTO BODY IF POSSIBLE TO AVOID TRAILING FLAP INDUCED SHOCK GENERATION

AERODYNAMICS

- o SPACE BASED AOTV'S THAT EXCEED LAUNCH VEHICLE ENVELOPE ARE NOT REQUIRED. CONFIGURATION TRENDS OF LOWER TOTAL SURFACE AREA (INDICATOR OF WEIGHT PENALTY) AND LOWER SURFACE TEMPERATURES (LIGHTER TPS) THAT LEAD TO AMOSS/BICONIC TYPE CONFIGURATIONS
- o RECOMMENDATION FOR ADVANCED LOX-H_2 ENGINES
  - TOTAL THRUST 8-13K LBSF
  - MAN RATED CARGO VEHICLE - 4-3000 LBF

PAYLOAD MANIFESTING

- o APPEARS TO BE SMALL ADVANTAGE TO LARGE (VS SMALL) AOTV
- o SPACE BASED AOTV SHOULD BE CAPABLE OF OPERATING FROM GROUND BASED MODE
- o AT THIS TIME, PROPELLANT FOR A CARGO TRANSPORT AOTV - SPACE OR GROUND BASED, SHOULD BE EARTH STORABLE N_2O_4-MMH

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3.0 PHASE II STUDY RESULTS

3.1 General Mission Model

The generalized mission model addressed is summarized in the Figure. The initial ground based AOTV's will be deployed from a 150 nmi circular orbit, launched from ETR at an orbital inclination of 28.5°. The space based AOTV's will be in a 200 nmi circular orbit at 28.5° inclination. Launch vehicles considered inclulde the standard STS, an improved STS, the aft cargo compartment (ACC) and the shuttle derived cargo vehicle.

AOTV GENERAL MISSION MODEL DEFINITION

LEO - LOW EARTH INITIAL PARKING ORBIT

- GROUND BASED 150 NM CIRCULAR, $i = 28.5°$
- SPACE BASED 220 NM CIRCULAR, $i = 28.5°$

HEO - HIGHER EARTH ORBIT DESTINATIONS

BASELINE:
- GEO - GEOSYNCHRONOUS EQUATORIAL ORBIT, $i = 0°$
- MOLNIYA - 12 HR PERIOD, $i = 63.4°$, $h_a = 21500$ NM, $h_p = 400$ NM

- SIX HOUR POLAR - ETR LAUNCH ~ 5600 NM CIRCULAR
- 5X GEO

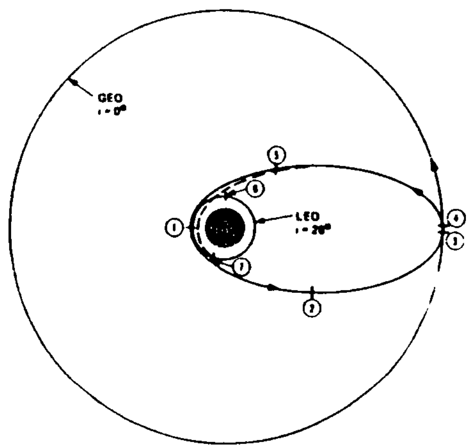
TIME CONSTRAINED MISSION - SINGLE ORBIT DEPLOYMENT, LOITER, AND RECOVERY

Operating scenarios were established for the several reference missions and ΔV budgets determined for use in the performance computations.

GEO delivery was examined in two modes. In a single stage mode (Figure A), an AOTV delivers the satellites to their final destination at GEO, then returnsto LEO. Three major propellant burns are required. In a two stage mode ("perigee kick", Figure B), the AOTV delivers the satellites to an elliptical GEO transfer orbit at 28^{o} inclination. The satellites' own on-board propulsion systems (the GEO station keeping systems, with enlarged propellant tankage) perform the GEO insertion burn (event 4, Figure B), while the AOTV coasts toward return to LEO. Only two small propellant consuming burns (events 6 and 8) are requried of the AOTV after payload separation. Effective use of aeromaneuvering orbital plane change during return from Molniya, Figure C, is not possible due to the large ΔV required for apsis rotation when leaving the Molniya orbit in order to place the in-atmosphere flight segment near the nodal crossing. Instead, it is recommended that the mid L/D AOTV be flown in the braking mode only with aeromaneuvering used for altitude control thus offering a substantial propellant reduction when compared with an all propulsive reusable OTV.

Figure A   Operating Scenario for AOTV-GEO Missions

A. Single Stage Transport



EVENTS

1 GEO TRANSFER BURN
ΔV_1 = 8090 FT SEC

2 MIDCOURSE CORRECTION
ΔV_2 = 50

3 CIRCULARIZATION AND INCLINATION CHANGE AT GEO
ΔV_3 = 8000

4 TRANSFER TO hp = 80 NM.
ΔV_4 = 5414 FOR L/D = 1.5
Δi AERO = 13°

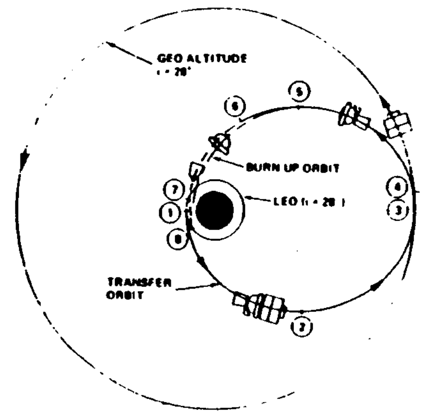
5 MIDCOURSE CORRECTION
ΔV_5 = 50

6 AOTV ENTERS ATMOSPHERE

7 LEO CIRCULARIZING AND PHASING BURNS
ΔV_7 = 450

Figure B   Operating Scenario for AOTV-GEO Missions

B. Two Stage Transport



EVENTS

1 GEO TRANSFER BURN
ΔV_1 = 8084 FT/SEC (PERIGEE KICK)

2 MIDCOURSE CORRECTION
ΔV_2 = 50 FT/SEC

3 STAGE PAYLOAD

4 PAYLOAD PERFORMS GEO CIRCULARIZING & PLANE CHANGE BURN
ΔV_4 = 5850 FT/SEC

5 STAGE DROP TANK
DROP TANK BURN
ΔV_5 = 50 FT/SEC

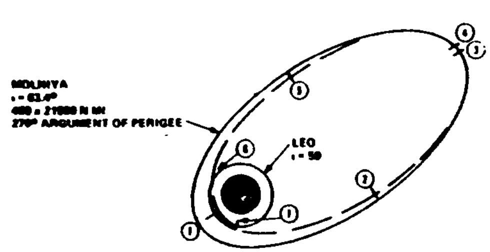
6 AOTV MIDCOURSE CORRECTION
ΔV_6 = 50 FT SEC

7 AOTV ENTERS ATMOSPHERE

8 AOTV LEO CIRCULARIZING & PHASING BURN
ΔV_8 = 500 FT/SEC

- AOTV Supplies perigee impulse
- Satellite (payload) supplies apogee impulse

Figure C   Operating Scenario for Molniya Missions



EVENTS

1 TRANSFER BURN
ΔV_1 = 8330 FT/SEC

2 MIDCOURSE CORRECTION
ΔV_2 = 50 FT SEC

3 PLANE CHANGE AND MOLNIYA INSERTION
ΔV_3 = 1170

4 TRANSFER TO hp = 80 N MI
WITH 35° APSIS ROTATION
AND 3.4° Δi
ΔV_4 = 8500
OR
WITH 5° APSIS ROTATION
AND 3.4° Δi
ΔV_4 = 1700

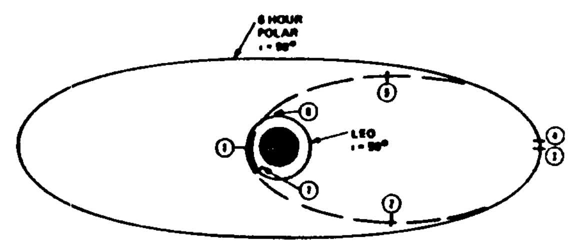
5 MID COURSE CORRECTION
ΔV_5 = 50

6 AOTV ENTERS ATMOSPHERE

7 LEO CIRCULARIZATION AND PHASING BURNS
ΔV_7 = 400

- MAXIMUM AERO PLANE CHANGE OBTAINED WHEN IN ATMOSPHERE FLIGHT IS AT NODAL CROSSING
- ΔV PENALTY TO ACCOMPLISH (APSIS ROTATION) IS PROHIBITIVE
- RECOMMEND LIFT VECTOR USED FOR FLIGHT CONTROL ONLY

Figure D   Operating Scenario for Polar Missions



EVENTS

1 TRANSFER BURN
ΔV_1 = 5400 FT/SEC

2 MIDCOURSE CORRECTION
ΔV_2 = 50

3 PLANE CHANGE AND CIRCULARIZATION AT 6 HR POLAR
ΔV_3 = 7000

4 TRANSFER TO hp = 100 N. MI.
ΔV_4 = [illegible] FOR L/D = 1.5
Δi AERO = 10°

5 TRANSFER hp = 80 N. MI.
ΔV_5 = 110

6 AOTV ENTERS ATMOSPHERE

7 LEO CIRCULARIZATION AND PHASING BURNS
ΔV_7 = 400

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In the Phase I Ground Based portion of this study, it was assumed that great flexibility existed on launch time so that there was only a small V budget for phasing. In this Space Based portion (Phae II), however, it is seen that lengthy departure delays are required due to the nodal regression rate differences unless a much larger Δ V budget is provided.

ΔV BUDGETS FOR SPACE BASED MISSIONS

- ALL PROPULSIVE ROUND TRIP -

- MANY SPACE BASED MISSIONS CAN BE IDENTIFIED WHERE LENGTHY DEPARTURE DELAYS (HUNDREDS OF DAYS) ARE REQUIRED DUE TO NODAL REGRESSION RATE DIFFERENCES, IF ΔV CAPABILITY = GROUND BASED RESULTS OF PHASE I.

SPACE STATION @ 220 N. MI I (°)	MISSION TO H/I (N.MI/°)	V_{MIN} (FT/SEC)	WAIT TIME (DAYS)	V_{MAX} (FT/SEC)
60	1400/60	6060	130	42000
28.5	GEO/28.5	25000	51	31000
28.5	GEO/0	PHASE I	0	PHASE I

3.2 Systems Analysis

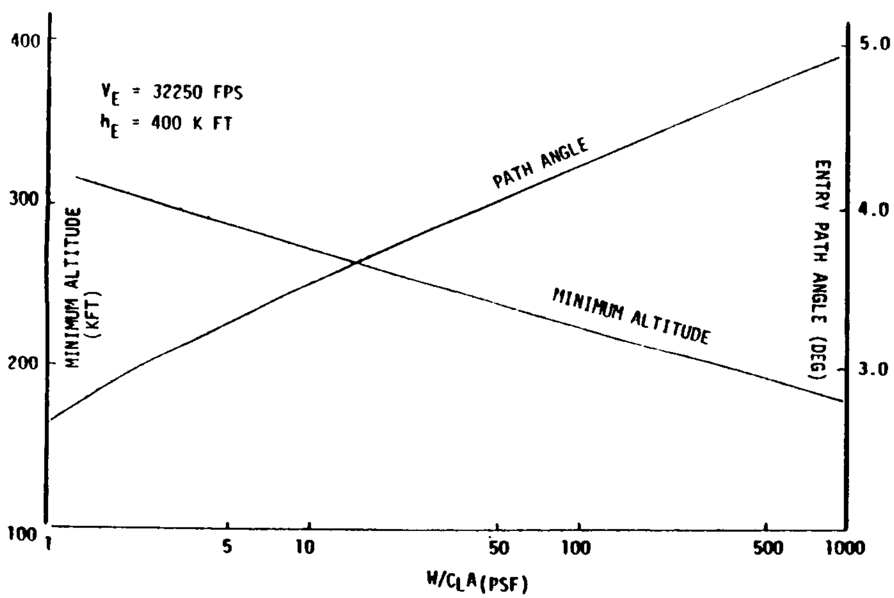
3.2.1 Aeromechanic Performance

Modulation of the lift vector for mid L/D AOTV can be to provide trajectory control (null accumulated errors) and to change the orbital plane (if desired). In other sections, it is demonstrated that large performance gains are realized by placing the apogee inject propulsion on the payload (AKP), and utilizing the AOTV only as a reusable perigee kick vehicle. In this no-plane-change mode, nearly all of the lift produced can be used to help "capture" in the one pass mode and thus the vehicle flies considerably higher, and hence cooler, than if the lift vector were being employed to change the inclination of the orbital plane. At the one pass overshoot bound, the minimum altitude, and hence maximum heat transfer, is driven only by the lift loading parameter W/C_LA. These variations are illustrated in the Figure.

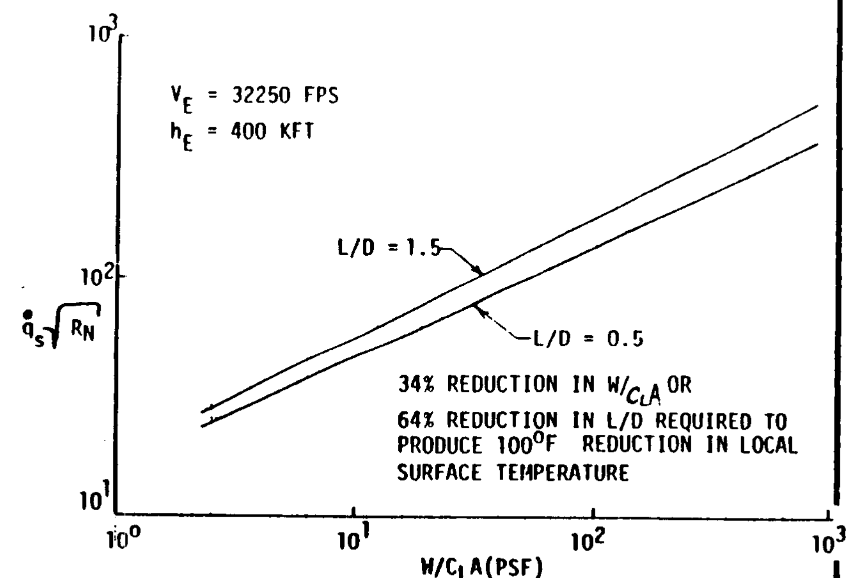
RE-ENTRY SYSTEMS OPERATIONS

GEO RETURN OVERSHOOT BOUND

MINIMUM ALTITUDE



MAXIMUM HEAT TRANSFER



34% REDUCTION IN W/C_LA OR
64% REDUCTION IN L/D REQUIRED TO PRODUCE 100°F REDUCTION IN LOCAL SURFACE TEMPERATURE

3.2.2 Space Based Configuration Development

Numerous configuration guidelines for Space Based mid L/D AOTVs are available from the ground based configurations developed in Phase I. The development history of the ground based configurations is outlined in the Figure.

GROUND BASED MID L/D AOTV CONFIGURATION HISTORY

- LMSC AMOOS + BAC L/D = .75 DATA BASE USED AS POINT OF DEPARTURE
- GE DATA BASE FROM STRATEGIC R/Vs EXTENDS TO $\theta_F = 4^\circ$
- MID L/D PHASE I GENERATED $\theta_F = 1^\circ$ & 2° ANALYTICAL DATA BASE
- GENERIC MID L/D AOTVs DEFINED
- AOTV PERFORMANCE DRIVEN BY (IN ORDER OF IMPORTANCE)
  - STAGING/DROP TANK SCENARIO - DELIVERY MISSIONS
  - ADVANCED TECHNOLOGY
  - L/D FOR LARGE PLANE CHANGE MISSIONS
  - PROPULSION MIXTURE RATIO & AFT FRUSTUM ANGLE

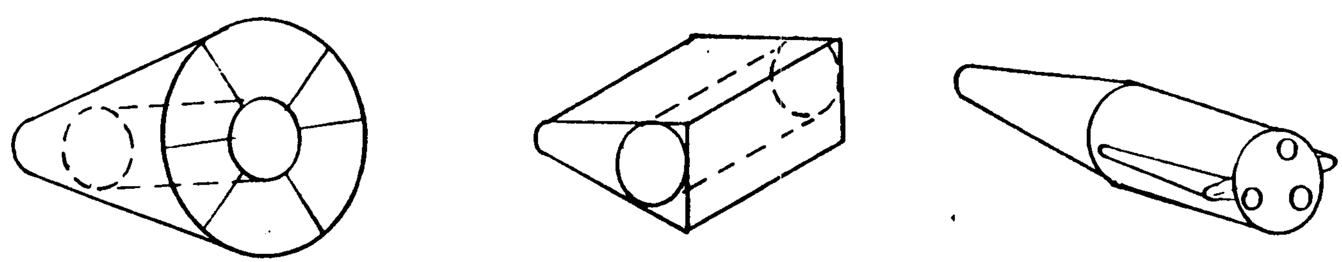
3.2.2.1 Alternate Mid L/D AOTV Space Based Configurations

The trends of higher flight altitude with decreased lift loading, $W/C_L A$, suggest exploration of alternate mid L/D AOTV configurations that might fly higher and hence, perhaps cooler. Cooler implies lighter TPS. Several types of configurations were evaluated, sphere cones, blunted wedges and a biconic similar to those evaluated previously in this study to which had been added Strakes. Each of these vehicles was sized to carry a propellant tank consistent with the delivery missions. The baseline TPS and structure weight scaling algorithms developed in Phase I were used to estimate total dry weight of the AOTV.

DEVELOPMENT OF SPACE BASED CONFIGURATIONS

THAT EXCEED LAUNCH VEHICLE ENVELOPE

- o FOR SPACE BASED AOTV, MODULES/SECTIONS CAN BE ASSEMBLED IN ORBIT
- o PEAK LOADS AND HEATING GENERALLY DECREASE WITH INCREASED MINIMUM ALTITUDE
- o MINIMUM ALTITUDE FOR ONE PASS CAPTURE AT OVERSHOOT BOUND IS DRIVEN BY LIFT LOADING PARAMETER W/C_LA
- o GEO DELIVERY PERIGEE KICK MID L/D AOTV'S CAN FLY AT OVERSHOOT BOUND
- o LARGE CONFIGURATIONS DEVELOPED FOR EVALUATION THAT HAVE $W/C_LA \ll W/C_LA$ BICONICS
- o SPHERE CONES, WEDGES, STRAKED BICONICS EVALUATED

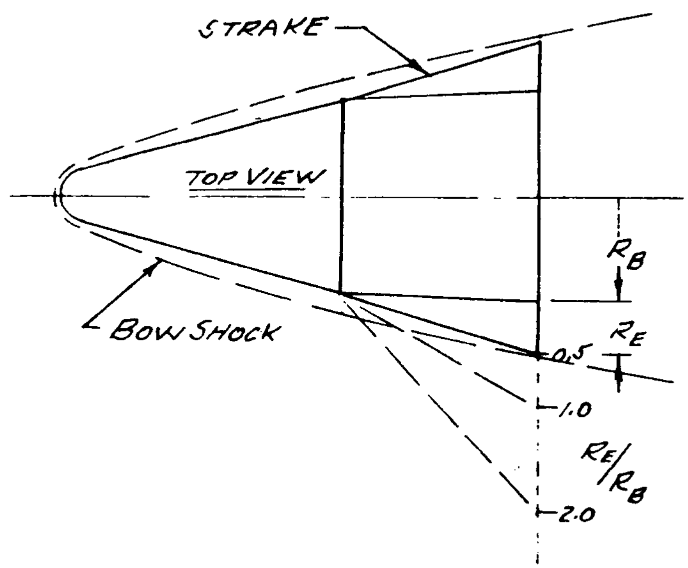


- o PROPELLANT TANK VOLUME SELECTED OF 1000 FT^3 (MR=7 GEO DELIVERY)
- o CYLINDRICAL TANK GEOMETRY EMPLOYED
- o TPS + STRUCTURE WEIGHT HELD AT BASELINE NUMBERS TO SCOPE W/C_LA

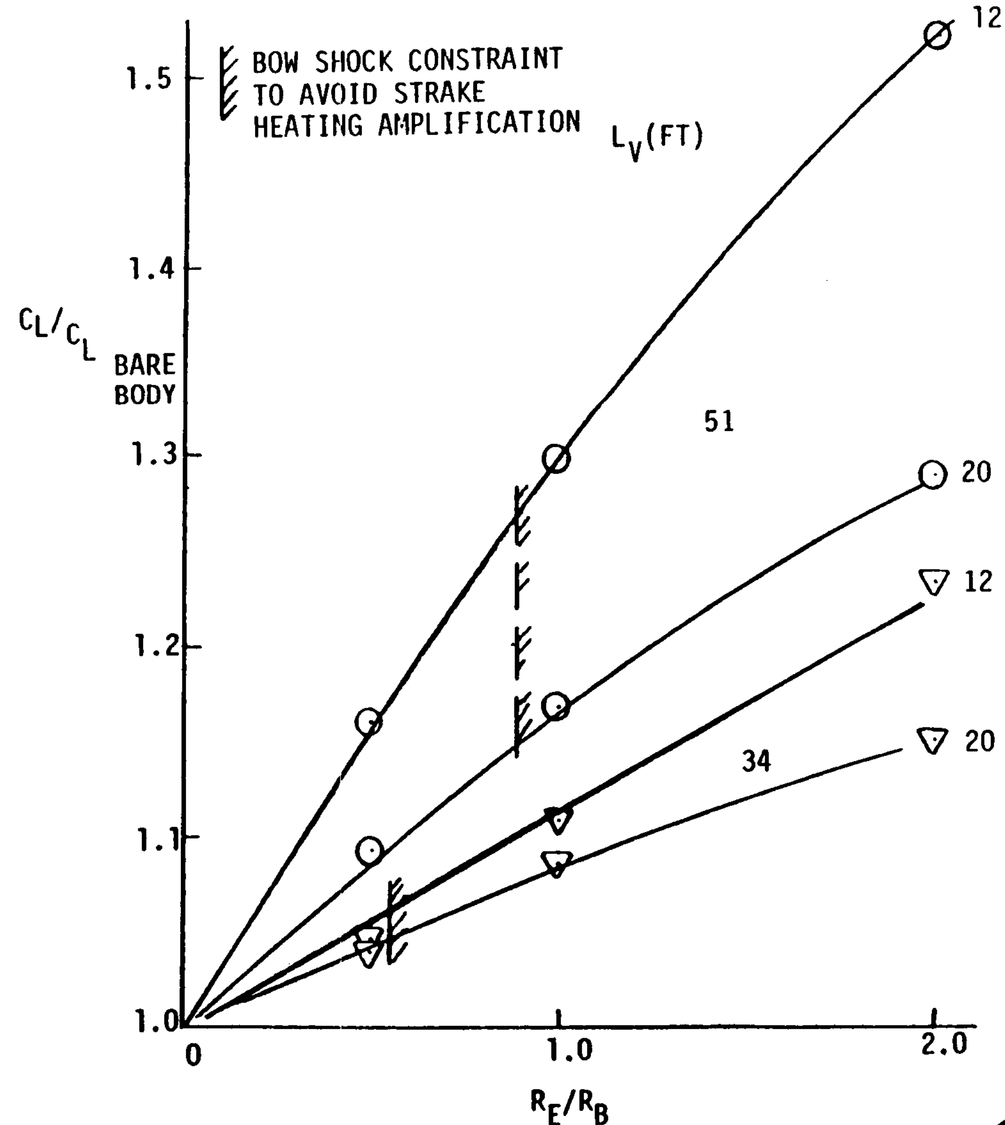
Substantial interest has developed in other programs of adding Strakes (skinny wings) to a cone or biconic vehicle. These Strakes can increase the lift generated by the AOTV at angle of attack when placed in the yaw plane, or can generate significant side force if placed in the pitch plane. In this attempt to reduce W/C_LA, they are located in the yaw plane and their span restricted to prevent bow shock impingement and subsequent heat transfer amplification that would produce significant leading edge thermal protection system problems.

Aerodynamic characteristics of a Straked biconic have been evaluated at angles of attack of 12 and 20^o for Strake span to base radius ratios, R_E/R_B, of up to 2.0. A review of the flow field bow shock geometry computations indicates that for the 34 foot delivery vehicle, the Strake span will be restricted to about 0.5 R_B, while for the 51 foot manned vehicle, the Strake span could be about 0.9 R_B. If the Strake span is restricted to these small values, it is seen in the Figure that lift augmentation is increased by 9% on the delivery vehicle and 20% on the manned vehicle. Since it takes a reduction of W/CL_A of about 34% to produce a surface temperature reduction of 100^{o}F, the addition of Strakes does not seem to be a fruitful approach.

STRAKED BICONIC



EFFECT OF STRAKE SPAN AND ANGLE-OF-ATTACK ON LIFT AUGMENTATION



STRAKE GENERATED LIFT AUGMENTATION IS LIMITED DUE TO BOW SHOCK CONSTRAINT

Based on previous sphere-cone-strake results, we can expect to get more ΔC_L and $\Delta L/D$ from larger strakes on biconics with frustum (rather than nose) dominated aerodynamics. The two biconics that have been arbitrarily selected for this study are considered to be nose dominated. The size of the strake can also be increased by increasing the AOTV nose radius thus driving the bow shock further off the frustum. Some loss of biconic L/D occurs due to the increased nose radius.

However, since local surface temperature is relatively insensitive to W/C_LA and L/D (34% reduction in W/C_LA produces only 100° decrease in surface temperature), it is recommended that this approach of adding strakes to the basic biconic not be pursued.

SUMMARY OF AERODYNAMIC CHARACTERISTICS FOR BICONIC MID L/D AOTVs WITH STRAKES

VEHICLE	BASIC VEHICLE L/D	BASIC VEHICLE W/C_DA	BASIC VEHICLE W/C_LA	LARGEST STRAKE THAT CAN BE ADDED R_E/R_B	$(L/D)/(L/D)_{BASIC}$	L/D	$(C_D)/(C_D)_{BASIC}$	$(C_L)/(C_L)_{BASIC}$
(4) 13.2 GEO DELIVERY	1.03	105	102	0.56	1.05	1.08	1.04	1.09
(10) 14 KLBS UP & BACK (4) + CREW MODULE + INT TANK	1.2	525	438	0.89	1.09	1.31	1.098	1.20

FROM OUR PARAMETRIC TRAJECTORY STUDIES

LOOKS LIKE $\dot{q} \sim \left(W/C_LA\right)^{.5}$

AND $\sim \left(L/D\right)^{0.2}$

SINCE TO GET 100°F TEMP REDUCTION REQUIES A 34% REDUCTION IN W/C_LA OR A

64% REDUCTION IN L/D

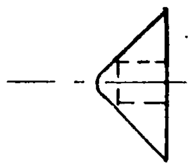
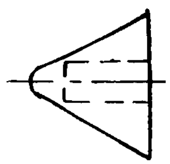
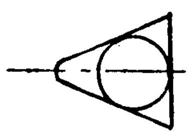
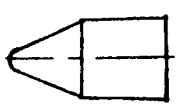
CONCLUDE: ADDING STRAKES TO BICONIC IS NOT REALISTIC WAY TO REDUCE TEMPERATURES

Additional configurations were explored in an attempt to develop attractive combinations of reduced lift loading, $W/C_L A$, and total surface area.

Blunted sphere cone and wedge configurations were evaluated and estimates of AOTV weight at entry was adjusted to reflect increase in surface area and hence increased structure and thermal protection subsystem weights.

For comparison, the biconic perigee kick GEO delivery vehicle has also been illustrated. Note that the trend of these larger vehicles is towards larger forebody surface areas, larger base areas, and higher local heat transfer rates in the areas that represent the major AOTV skirt areas. Even though some of the vehicles evaluated fly at considerably higher altitude, due to the relatively large cone angles and angle of attack, the local heat transfer rates are higher than those on the aft skirt of the biconic vehicle.

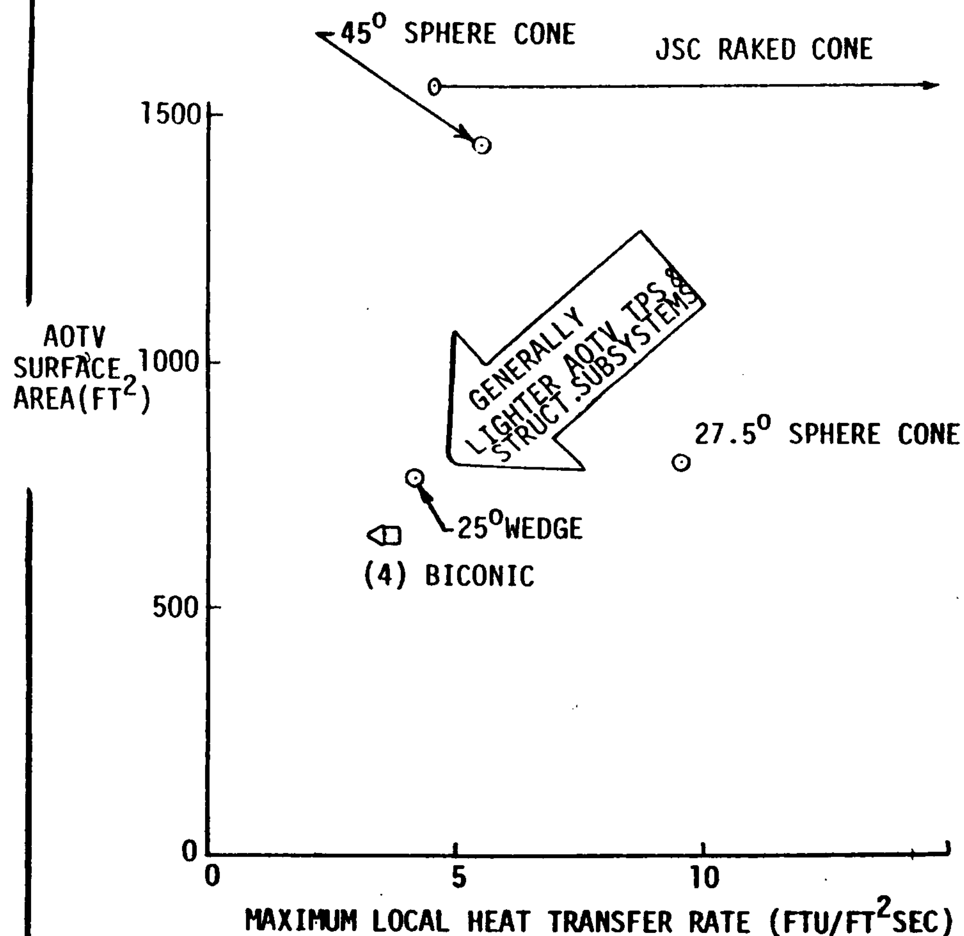
SUMMARY OF BLUNT CONE & WEDGE CONFIGURATIONS EVALUATED

CONFIGURATION		POTENTIAL L/D	FOREBODY SURFACE AREA(FT^2)	BASE AREA (FT^2)	MINIMUM FLT ALTD. (KFT)	W/C_DA (PSF)	$\dot{q}$ STAG MAX BTU/FT^2SEC	$\dot{q}_c$ MAJOR SKIRT AREA
	SPHERE/CONE $\theta_C=45^o$ $R_N/R_B=0.32$	0.2	1440	960	256	25	25	[illegible]
	SPHERE/CONE $\theta_C=27.5$ $R_N/R_B=0.2$	0.5	780	380	242	49	67	[illegible]
	WEDGE $\theta_W=25^o$ $\frac{R_N}{R_B}=0.2$ L=20' $R_T=4'$	1.4	760	160	274	10.5	52	4.2
	BICONIC $L_V=28'$	1.0	650	113	223	123	142	[illegible]

A significant fraction of the AOTV dry mass is thermal protection system and structure. Minimization of AOTV dry mass generally provides lower recurring costs. Local thermal protection system mass is driven by the magnitude of the local heat transfer rate. Higher local heat transfer rates usually require greater local TPS mass for survival. The configuration evaluation results from the previous page are illustrated here. Note that the low heat transfer and the small surface area make the biconic mid L/D AOTV a very attractive configuration. This vehicle does experience high heat transfer in the nose region, but the area is quite small and several contemporary heat protection approaches are viable here.

RE-ENTRY SYSTEMS OPERATIONS

SPACE BASED AOTV CONFIGURATION CONCLUSIONS



- o SMALLER SURFACE AREA MINIMIZES TPS + STRUCTURE WEIGHT
- o LOWER HEAT TRANSFER ON MAJOR ACREAGE IS GOOD
- o FLIGHT PROVEN AERODYNAMICS

↓

IMPROVED PERFORMANCE

↓

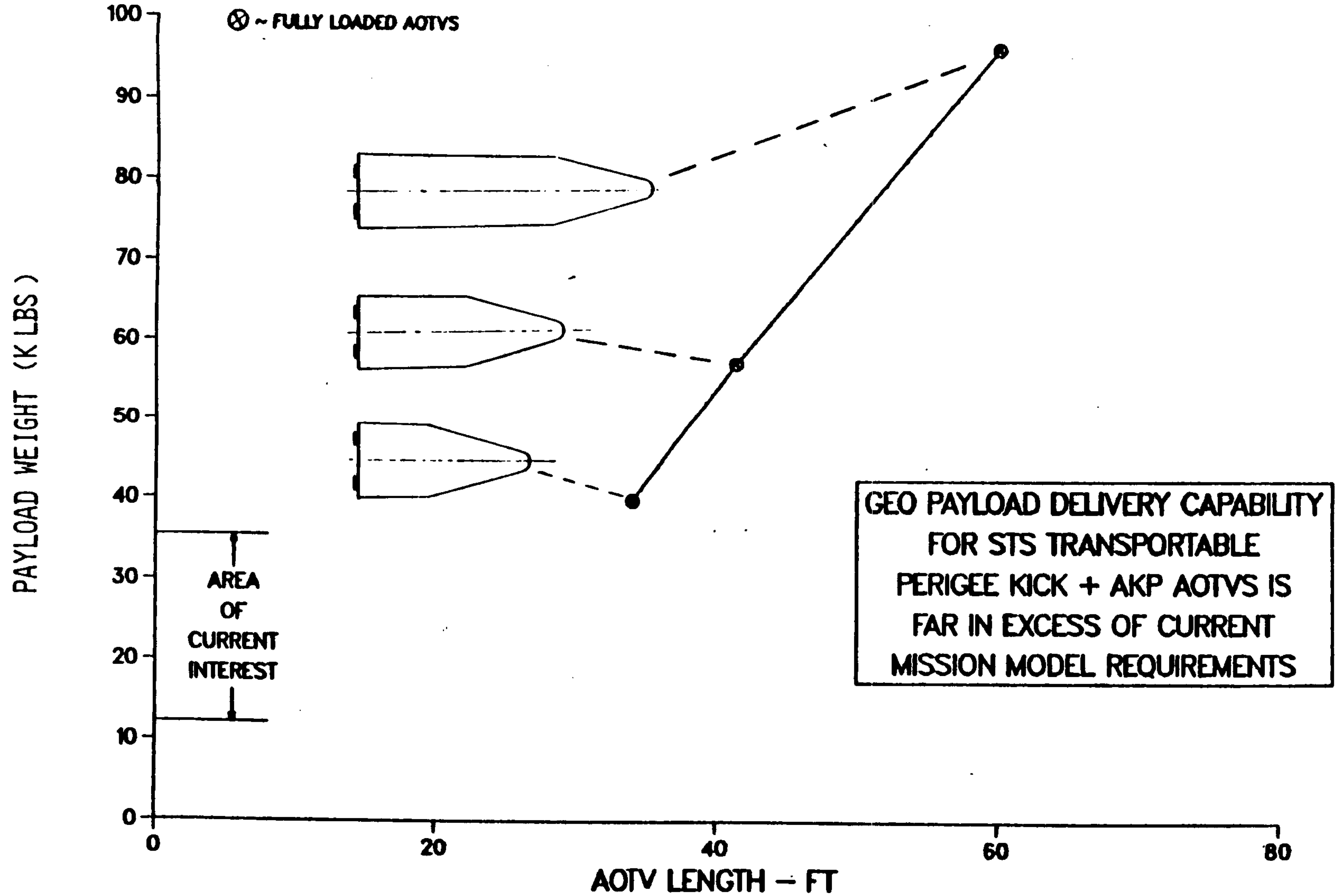
LOWER COST

BICONIC CLASS IS BEST
FOR SPACE BASED
MID L/D AOTV

3.2.2.2 Mid L/D AOTV Biconic Configuration Trending Analyses

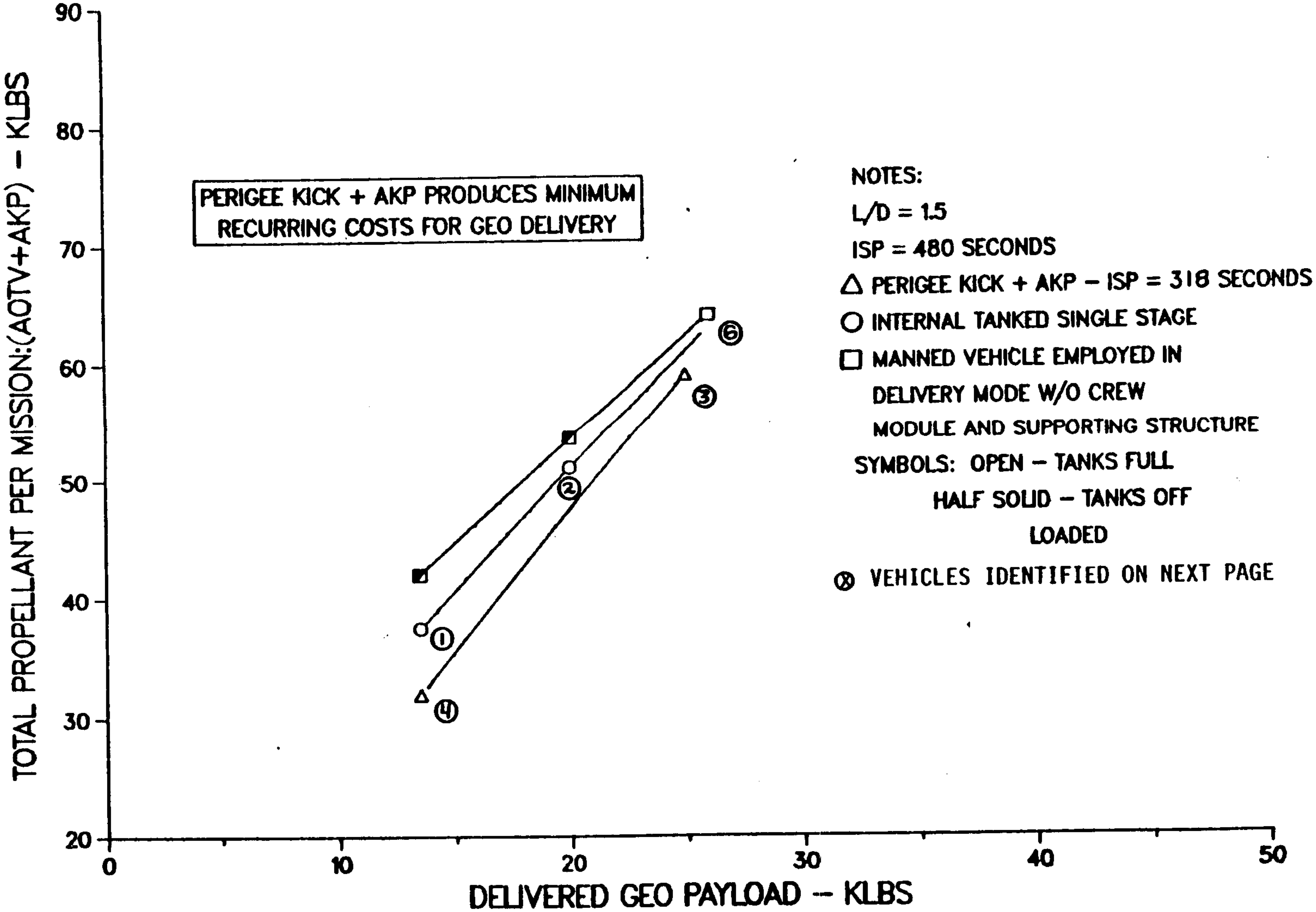
Employing the highly efficient perigee kick mode plus apogee kick propulsion (AKP), three vehicles were evaluated that varied in length from 34 to 58 ft. It was concluded that GEO payloads ranging in size from 40,000 to 90,000 lb could be delivered with these AOTV's. This range of payload sizes was far in excess of the typical "average" size that one would manifest considering the current mission model. It was concluded that much smaller AOTV's would provide greater utility across the mission model. The AKP employed in these studies is a scaled version of that used on Intelsat VI with storable N_2O_4-MMH propellants operating at a state-of-art I_{SP} = 318 sec.

SPACED BASED PERIGEE KICK + AKP MID L/D AOTVS
USING CRYOGENIC PROPELLANTS
ISP = 480 SECONDS



The concept of a universal man rates delivery vehicle was explored in our trending analysis and compared to 1) a baseline single stage delivery vehicle and 2) a perigee kick + AKP delivery vehicle. The man rated delivery vehicle consisted of a main propellant capability consistent with a manned round trip mission, with the crew module and supporting substructure removed, and the propellant tanks off loaded for delivery missions. The single stage internal tanked vehicles consisted of two discrete designs sized to deliver 13.2 and 20 K lb payloads. The perigee kick vehicles consisted of one of the single stage vehicles employed in the perigee kick mode and a perigee kick vehicle uniquely sized for delivery to GEO of a 13.2 K lb payload. Specific characteristics of these vehicles are compared in the figure. Propellant required plus AKP weight (if present) is considered the figure of merit, relating to the eventual total cost. Note that the perigee kick AOTV's offer a substantial advantage.

TOTAL PROPELLANT REQUIREMENT FOR SEVERAL MID L/D AOTV GEO DELIVERY MISSIONS



Specific characteristics of the space based mid L/D AOTVs that were compared in the previous figure, are summarized in the following table.

SPACE BASED MID L/D AOTVs USING CRYOGENIC PROPELLANTS

I_{SP} = 480 SEC L/D = 1.5

VEHICLE	W_{DRY}	$W_{P/L}$ (KLBS)	W_P (KLBS)	L_V (FT)
(1) 13.2K DELIVERY SINGLE STAGE	5114	13.2	37.5	34
(2) 20K DELIVERY SINGLE STAGE	5768	20.0	50.5	37.5
(3) (1) USED AS PERIGEE KICK + AKP	5114	24.9	37.5+AKP=58.6	34
(4) 13.2K DELIVERY PERIGEE KICK + AKP	3913	13.2	21.0+AKP=32.2	28
(5) 14K UP & BACK SINGLE STAGE	7740	14 UP 14 BACK	64.5	51
(6) (5) W/O CREW MODULE AND SUPPORTING STRUCTURE	6700	26.8	64.5	41.6
(7) (1) + DROP TANKS + CREW MODULE & SUPPORT STRUCTURE	7026	14 UP 14 BACK	62.0	44
(8) 8K UP & 6K BACK SINGLE STAGE	6080	8 UP 6 BACK	39.0	45
(9) (4) + DROP TANKS + CREW MODULE & SUPPORT STRUCTURE	6422	14 UP 14 BACK	61.0	38
(10) (4) + CREW MODULE + INTERNAL TANK	7247	14 UP 14 BACK	63.0	55

A dramatic comparison can be made in the trending analysis of the propellant transport cost advantage for a perigee kick plus AKP AOTV delivery vehicle. Considering 10 delivery missions per year over a 10 year period, with the transport costs to low earth orbit of $1000 per pound, it is seen that the difference between an off-loaded manned delivery vehicle and the more efficient perigee kick vehicle is one billion dollars. This clearly emphasizes the advantage of the perigee kick + AKP delivery option.

SPACE BASED MID L/D CRYOGENIC PROPELLANT TRANSPORT COSTS FOR SEVERAL DELIVERY SCENARIOS TO GEO

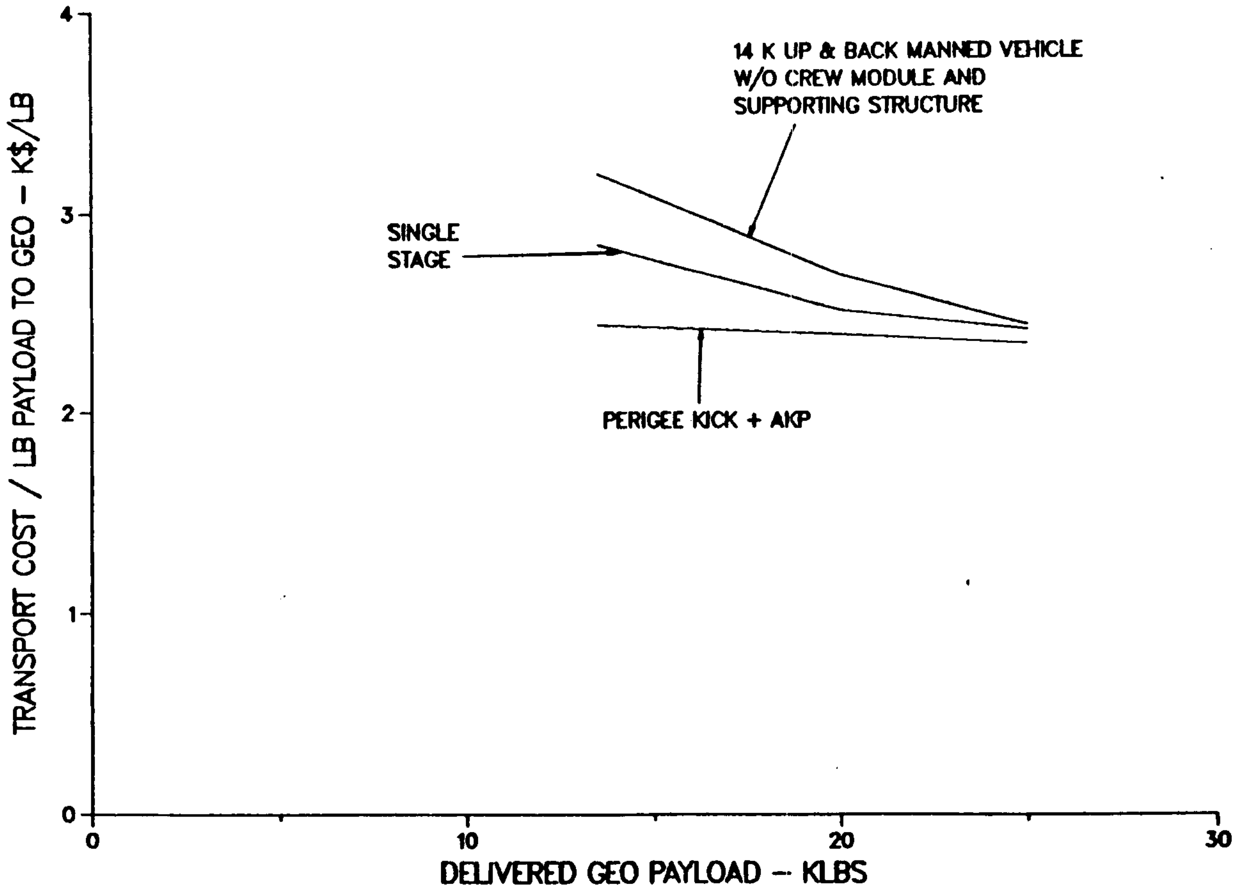
I_{SP} = 480 SEC
W_P = 13.2 KLBS
10 FLIGHTS/YR X 10 YRS
DELIVERY COST TO LEO = $1000/LB

VEHICLE	W_P (+ AKP)(KLBS)	TOTAL PROPELLANT TRANSPORT COST	COST/LB PAYLOAD TO GEO
(1) SINGLE STAGE	37.5	$3.75B	$2840
(5) 14K UP & BACK MANNED OFFLOADED VEHICLE W/O CREW MODULE OR SUPPORTING STRUCTURE	42.0	$4.2B	$3182
(4) PERIGEE KICK + AKP	32.2	$3.2B	$2440

- PROPELLANT TRANSPORT COSTS FOR GEO DELIVERY OF REPRESENTATIVE PAYLOADS ARE SIGNIFICANTLY LESS FOR PERIGEE KICK + AKP
- USE OF A STRIPPED MANNED VEHICLE OFFLOADED FOR GEO DELIVERY INCURS A $1B Δ IN PROPELLANT TRANSPORT COST

The trending analyses of the previous figure has been extended to determine the cost trends as the GEO payload size is increased from 13,200 lbs to 25,000 lbs. Note that the large differences in specific propellant transport cost that exists for the smaller payloads disappears as the GEO payload size is increased to 25,000 lbs. This is much larger than the typical payload in the current mission model.

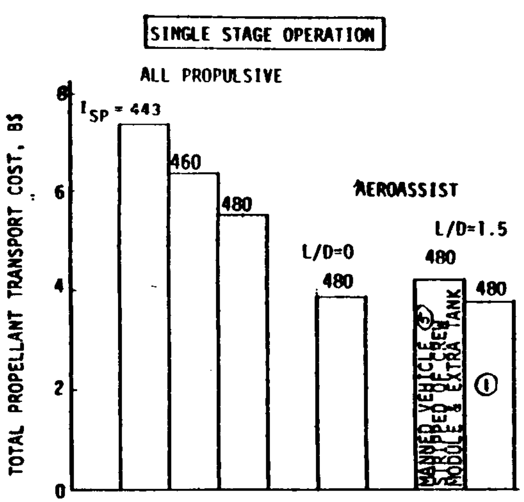
EFFECT OF DELIVERED PAYLOAD MASS ON SPECIFIC PROPELLANT TRANSPORT COST FOR THREE OPERATIONAL METHODS

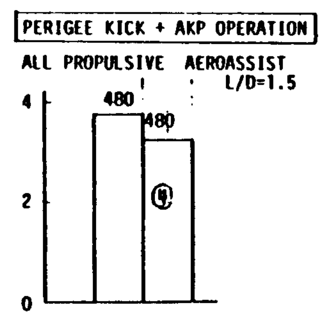


It is instructive to compare the mid L/D AOTV vehicles to all propulsive vehicles. All propulsive stages have been created by removing the TPS and nose fairing from the mid L/D vehicles. Illustrated here in the figure in trending analyses is the total propellant transport cost for all propulsive OTV's with state-of-the-art cryofueled I_{SP} = 443 sec and advanced technology versions with I_{SP} = 460 and 480 sec. Also compared in this all delivery scenario is the advantage of using the hypersonic L/D for orbital plane change and the advantage of using a perigee kick scenario in contrast to a single stage operation. Note that aeroassist provides a clear operational cost advantage over the all propulsive OTV; perigee kick + AKP provides a clear advantage over single stage operation; and use of a stripped manned vehicle off-loaded for GEO delivery incurs substantial penalty.

SPACE BASED CRYOGENIC PROPELLANT TRANSPORT COSTS FOR REUSABLE OTV GEO DELIVERY

W_{PL} = 13.2 K LBS
10 FLIGHTS/YR X 10 YRS
DELIVERY COST TO LEO = $1000/LB





- o AEROASSIST PROVIDES CLEAR OPERATIONAL COST ADVANTAGE OVER ALL PROPULSIVE OTV
- o PERIGEE KICK + AKP AOTV PROVIDES ADVANTAGE OVER SINGLE STAGE AOTV
- o USE OF STRIPPED MANNED VEHICLE OFFLOADED FOR GEO DELIVERY INCURS SUBSTANTIAL PENALTY

	VEHICLE CONCEPT	I_{SP}(SEC)	W_P(+AKP)(KLB)	TOTAL PROPELLANT TRANSPORT COST (\$)
DELIVERY — 10 FLIGHTS/YR X 10 YRS	ALL PROPULSIVE			
	SINGLE STAGE	443	73.0	7.30
	"	460	63.5	6.35
	"	480	54.9	5.49
	PERIGEE KICK+AKP	443	40.7	4.07
	"	480	37.5	3.75
	AEROASSIST			
	DERIVATIVE OF (1) OPERATED @ L/D=0 SINGLE STAGE	443	47.0	4.70
	"	480	38.7	3.87
	(5) 14K UP & BACK MANNED OFFLOADED VEHICLE W/O CREW MODULE OR SUPPORTING STRUCTURE	480	42.0	4.2
	(1) SINGLE STAGE,L/D=1.5	480	37.5	3.75
	(4) PERIGEE KICK+AKP	480	32.2	3.22
14 UP & BACK — 2 FLTS/YR X 10 YR	(10)(4)+CREW MODULE + INT TANK	480	63.0	1.26
	(10) IN L/D=0 MODE		68.7	1.37

The propellant transport costs illustrated in the table have been evaluated to determine the incremental cost advantage of several different uses of aeroassist or advanced technology. These cost advantages are summarized in the Table. Note the clear indication that introduction of aeroassist to a current technology engine provides a larger cost impact than introduction of an advanced technology engine in an all propulsive OTV. Numerous other interesting comparisons have been made and are illustrated.

SPACE BASED MID L/D AOTV SYSTEM PAYOFF

$W_{P/L}$ = 13.2 KLBS

10 FLTS/YR X 10 YRS

DELIVERY COST TO LEO = $1000/LB

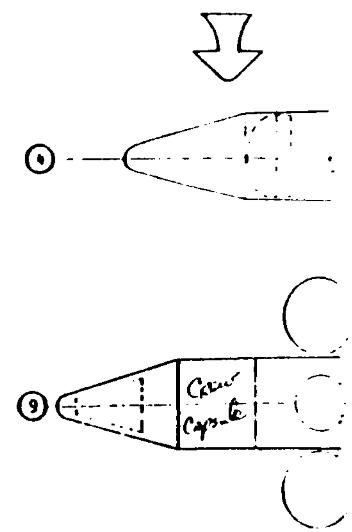
	COST SAVINGS (B$)
(1) ADD NEW ENGINE AND AEROASSIST (OPERATE @ L/D = 1.5) TO ALL PROPULSIVE SINGLE STAGE (I_{SP} = 443)	3.55
(2) ADD AEROASSIST (OPERATE MID L/D @ L/D = 0) TO ALL PROPULSIVE SINGLE STAGE (I_{SP} = 443)	2.6
(3) ADD NEW ENGINE (I_{SP} = 480) TO ALL PROPULSIVE SINGLE STAGE (I_{SP} = 443)	1.8
(4) ADD AEROASSIST (OPERATE MID L/D @ L/D = 1.5) TO ALL PROPULSIVE SINGLE STAGE (I_{SP} = 480)	1.6
(5) ADD NEW ENGINE TO SINGLE STAGE MID L/D AOTV (OPERATE @ L/D = 0)	0.8
(6) ADD AEROASSIST (OPERATE @ L/D = 0) TO ALL PROPULSIVE PERIGEE KICK STAGE (I_{SP} = 480)	0.53
(7) ADD NEW ENGINE (I_{SP} = 480) TO ALL PROPULSIVE PERIGEE KICK STAGE (I_{SP} = 443)	0.32

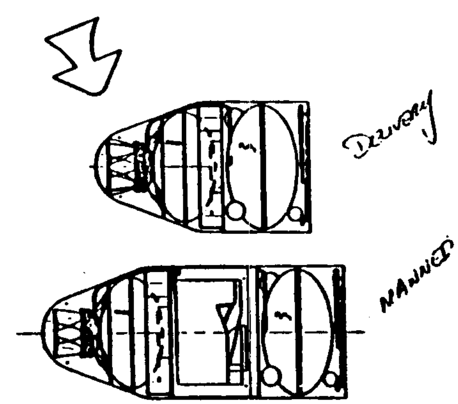
Two main issues continue to be investigated/debated. The real cost of transport to low earth orbit depends on the advantage that can be taken of space available on the regular shuttle missions to reduce the real propellant transport cost. Some missions, namely the manned missions, benefit from the utilization of propellant drop tanks that need not be covered with external thermal protection. It is not clear at this time if this mode is acceptable from the accumulating debris standpoint. The mid L/D AOTV configuration preferences that one might select, given various answers to the two major questions identified above, are illustrated in the figure.

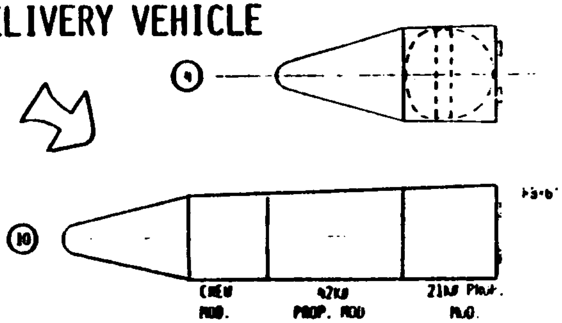
MID L/D AOTV CONFIGURATION SELECTION LOGIC

ARE THROW AWAY DROP TANKS ACCEPTABLE?

- YES → IS PROPELLANT TRANSPORT COST TO LEO EXPENSIVE
  - YES → DELIVERY: PERIGEE KICK STAGE + AKP, SIZE SELECTED TO HANDLE MOST OF-MISSION MODEL
    MANNED ROUND TRIP: PERIGEE KICK STAGE (FROM ABOVE) + CAPSULE SECTION + DROP TANKS
  - NO → PERFORMANCE IS NOT A DRIVER: USE MOST DENSE PROPELLANT INTERNAL TANKED VEHICLE SIZED FOR 14K UP & BACK AND OFF LOADED FOR DELIVERY MISSIONS OR USE CURRENT CRYO TECHNOLOGY, I.E., RL-10
- NO → IS PROPELLANT TRANSPORT COST TO LEO EXPENSIVE?
  - NO → PERFORMANCE IS NOT A DRIVER (see above)
  - YES → DELIVERY: PERIGEE KICK STAGE + AKP, SIZE SELECTED TO HANDLE MOST OF MISSION MODEL
    MANNED ROUND TRIP: ADD ANOTHER TANK SECTION PLUS CREW MODULE TO ABOVE DELIVERY VEHICLE





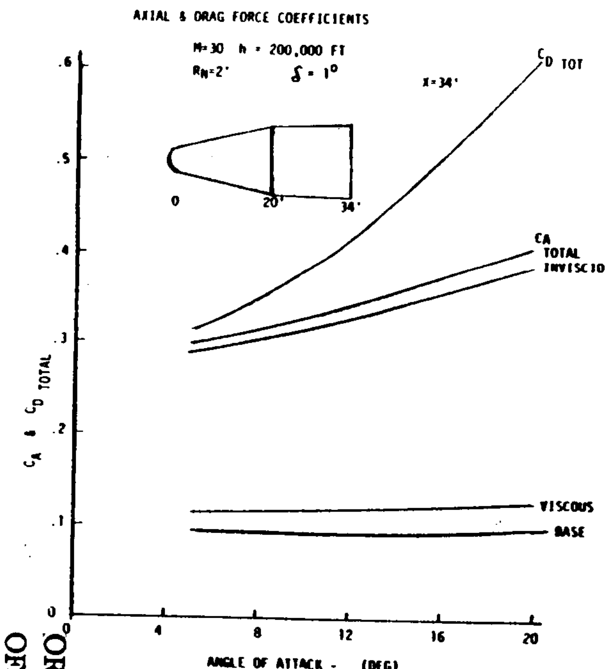


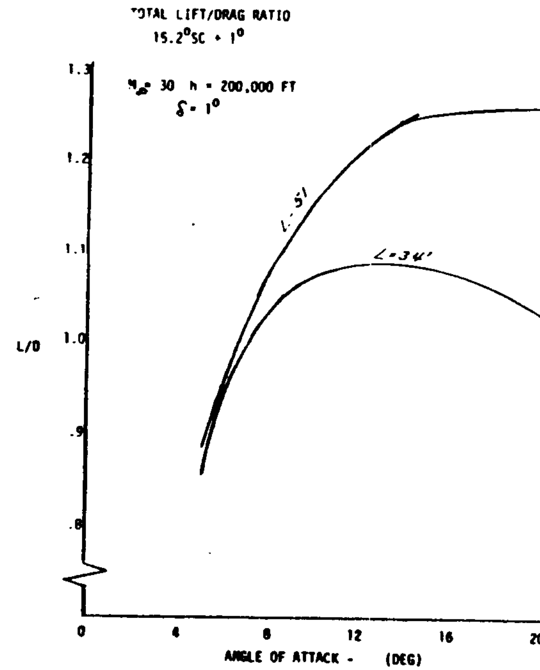
The modular configurations (4) and (10) were selected for the GEO delivery and manned round trip missions. Simultaneously with this selection process, aerodynamic characteristics were generated for short and long mid L/D AOTVs. The configurations selected were (1) and (7). The configurations were geometrically modified slightly to reduce the number of cases to be run.

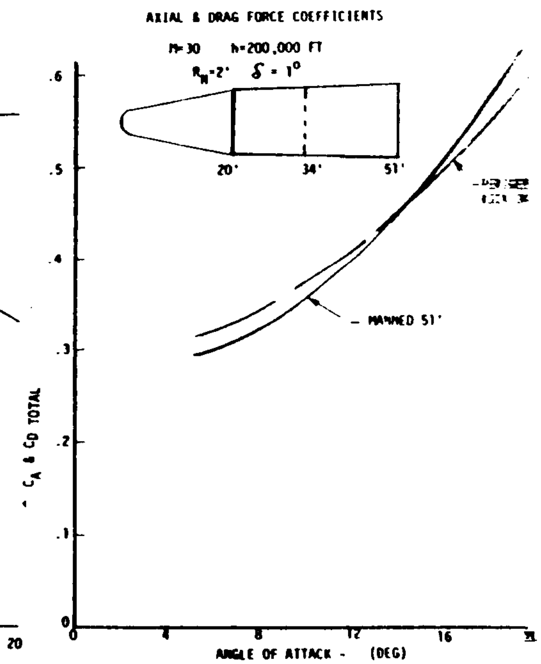
Results of the inviscid and laminar viscous computations are summarized in the figures. The drag coefficient increments are illustrated. The lift to drag ratios for these vehicles are compared in the middle figure. Note that the L/D variation with angle of attack has a broad top. This provides capability of operating at a X2 difference on C_D or C_L. Computations have been performed employing the 3DFF and 3VFF operational codes (AFFDL-TR-78-67 and SAMSO-TR-79-5).

RE-ENTRY SYSTEMS OPERATIONS

AERODYNAMIC CHARACTERISTICS OF DELIVERY AND ROUND TRIP VEHICLES







With the increased SDI activity, it would appear that projected missions exist that may justify, on the basis of cost, a program unique AOTV.

RE-ENTRY SYSTEMS OPERATIONS

ARGUMENT FOR PROGRAM UNIQUE AOTVs

- RECENTLY WE RESPONDED TO AN SDI RFP ON TECHNOLOGY BENEFITS
- MANY HEAVY SATELLITES PLACED AT NEAR EARTH HIGH INCLINATION ORBITS
- LIFE CYCLE COSTS DRIVEN BY LAUNCH COSTS
- A REUSABLE PROGRAM UNIQUE AOTV SAVED BILLIONS OF $

3.2.2.3 Evolutionary Capabilities of Space Based AOTVs

Work performed during Phase 1 of this contract identified a number of attractive AOTVs, each designed for a unique mission. The following set of figures display how two of these vehicles can perform very significant additional missions by replacing an original module with a different one. In the case of the small perigee kick cargo carrying OH-3, drop tanks for standard satellite delivery missions can be replaced by very large external tanks (either drop tanks or reuseable tanks), increasing its cargo capacity by a factor of 4 (Figures 02 to 05). In the case of H-1M, removing the bare bones capsule (for servicing work at GEO) allows installation of a capsule for transporting 5 or 6 people between LEO and GEO (Figures 012 to 016).

Another form of evolution is displayed in Figures 06 to 011. This evolution utilizes a pre-existing cargo carrying AOTV (OH-3) as the first stage of a two-stage manned GEO mission. The second (manned) stage of this system is designated OH-2M.

EVOLUTIONARY CAPABILITIES OF SPACE BASED AOTVs

- OH-3
  - HEAVY PAYLOAD PLACEMENT
  - MANNED GEO MISSION WITH OH-2M
- H-1M
  - 5 OR 6 PERSON TRANSPORTATION TO GEO

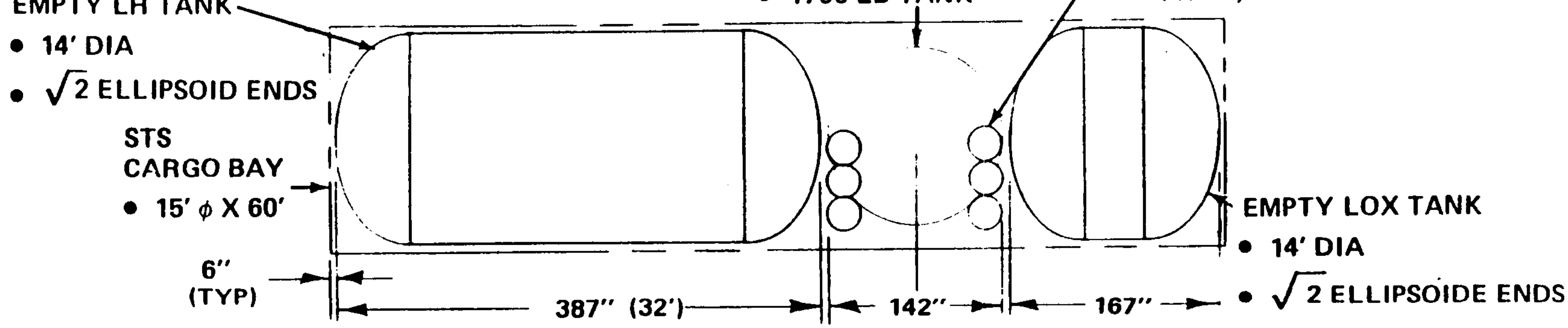
V84-1106-009(T)

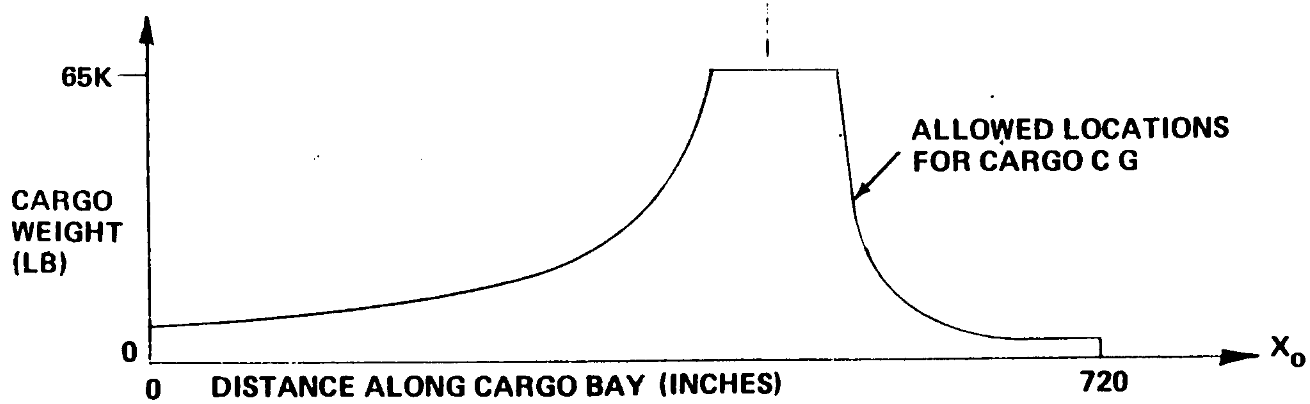
The cargo bay packaging of a filled Hydrazine tank and large, empty cryogen tankage sized for the delivery of 50,000 lbm of hydrazine and 1400 lbm of nitrogen to GEO is illustrated. The package as shown meets the length, weight and CM requirements of the STS when loaded with cargo (hydrazine and nitrogen) and with the propellant tanks empty. The empty LO_2/LH_2 propellant tanks are sized for the propellant capacity required by the $OH-3^2$ ($3-LO_2/LH_2$) for delivery of the 56250 lbm payload to GEO in a single perigee burn single stage insertion mission. Cryogenic propellant for the empty tanks is supplied by a space station depot.

SPACE BASED OH-3: DELIVER 50,000 LB HYDRAZINE TO GEO

IRAD

- SINGLE 65K SHUTTLE LAUNCH
- DELIVER 55,300 LB PAYLOAD TO GEO
- PROPELLANT SUPPLIED BY LEO DEPOT
  - O/F = 6:1

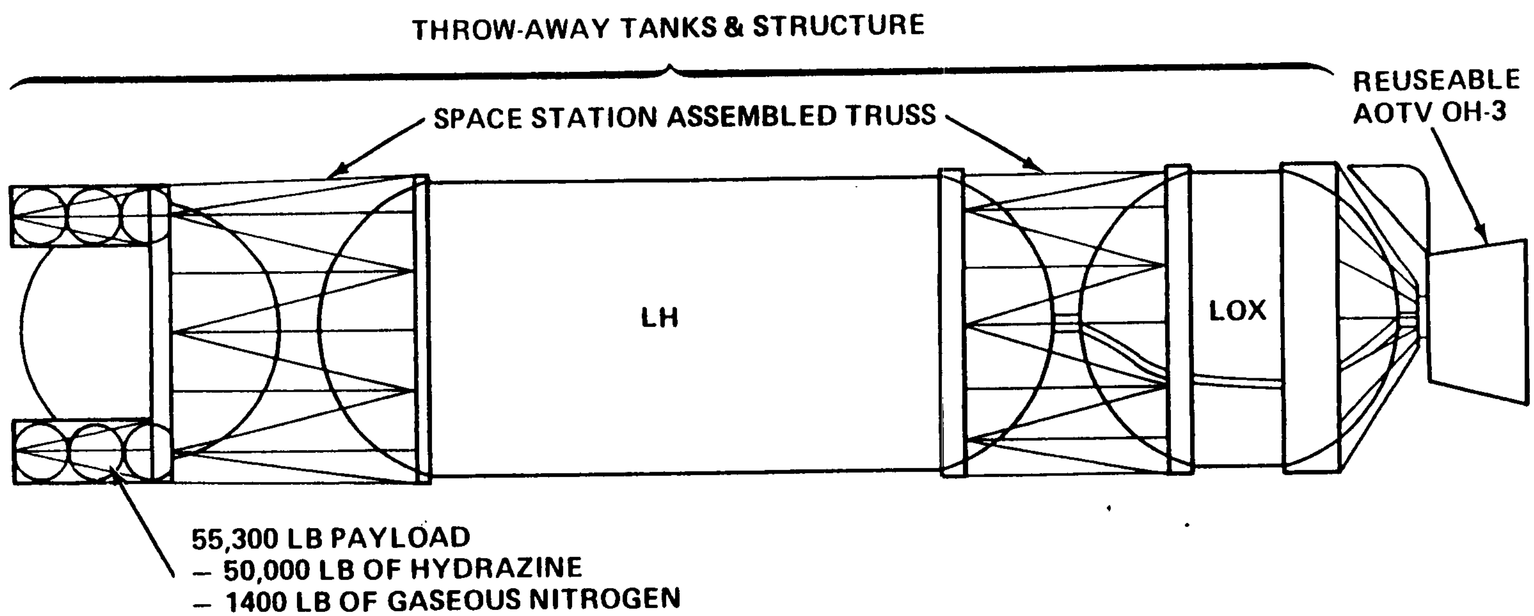




V83-2102-006G

The facing figure illustrates the space based OH-3 in flight configuration, with space station assembled truss structures at the interfaces of the OH-3/oxidizer tank, oxidizer tank/fuel tank, and fuel tank/GEO payload. All the non-GEO equipment shown, with the exception of the reusable AOTV OH-3, is disposable. The package delivers 55,300 lbm of payload to GEO in a single perigee burn, single stage insertion. 51,400 lbm of the delivered payload is usable propellant for satellites at GEO.

SPACE BASED OH-3: LARGE HYDRAZINE DELIVERY TO GEO



GRUMMAN

V84-1106-010(T)

The performance and mass characteristics of OH-3 in this heavy payload delivery mode is shown. Note that very large gravity losses (ΔV = 881 ft/sec) are associated with this low thrust/weight, single perigee burn mission. As this payload and AOTV are insensitive to Van Allen Belt radiation, a multiple perigee burn mission is easily obtained. Multiple perigee burns could reduce gravity losses to inconsequential levels, with a resulting significant increase in payload, or, not disposing the external tanks and truss structures. This last can be achieved by flying this infrequent mission in an all propulsive mode. Since the AOTV will not enter the atmosphere, the tanks and structure can be retained for future use.

C-2

PERFORMANCE OF OH-3SB: GEO HYDRAZINE DELIVERY (1-1/2 STAGE)

IRAD

LEO CAPABILITY	65,000 LB
ASE	UP TO 1770 LB
PAYLOAD DELIVERED	55,300 LB
PAYLOAD RETURNED	0
DROP TANK "1/2 STAGE"	7930 LB
AOTV "DRY" WEIGHT	4000 LB
AOTV ATMO ENTRY	4150 LB
PROPELLANT:	132,230 LB
DROP TANK: USEABLE	(129,590)
DROP TANK: RESERVES & RESIDUALS	(730)
DROP TANK: INFLIGHT LOSSES	(1760)
AOTV INTERNAL	(150)
GLOW	199,460 LB

I_{SP} = 475 SEC, THRUST = 12,000 LB

$(T/W)_1 = 0.06 \longrightarrow$ GRAVITY LOSS = 881 FT/SEC

$(T/W)_4 = 2.9 \longrightarrow$ GRAVITY LOSS $\approx$ 0

ΔV_1 = 8860 FT/SEC

ΔV_2 = 6000 FT/SEC

$\Delta V_3 = 5200$ FT/SEC $\longrightarrow \Delta i_{AREO} = 15^\circ$

$(L/D)_{REQD} \geq 1.04$

ΔV_4 = 500 FT/SEC

SPACE BASED AOTV + PROPELLANT DEPOT AT LEO INCREASES GEO DELIVERY 5X WITH AOTV REUSE

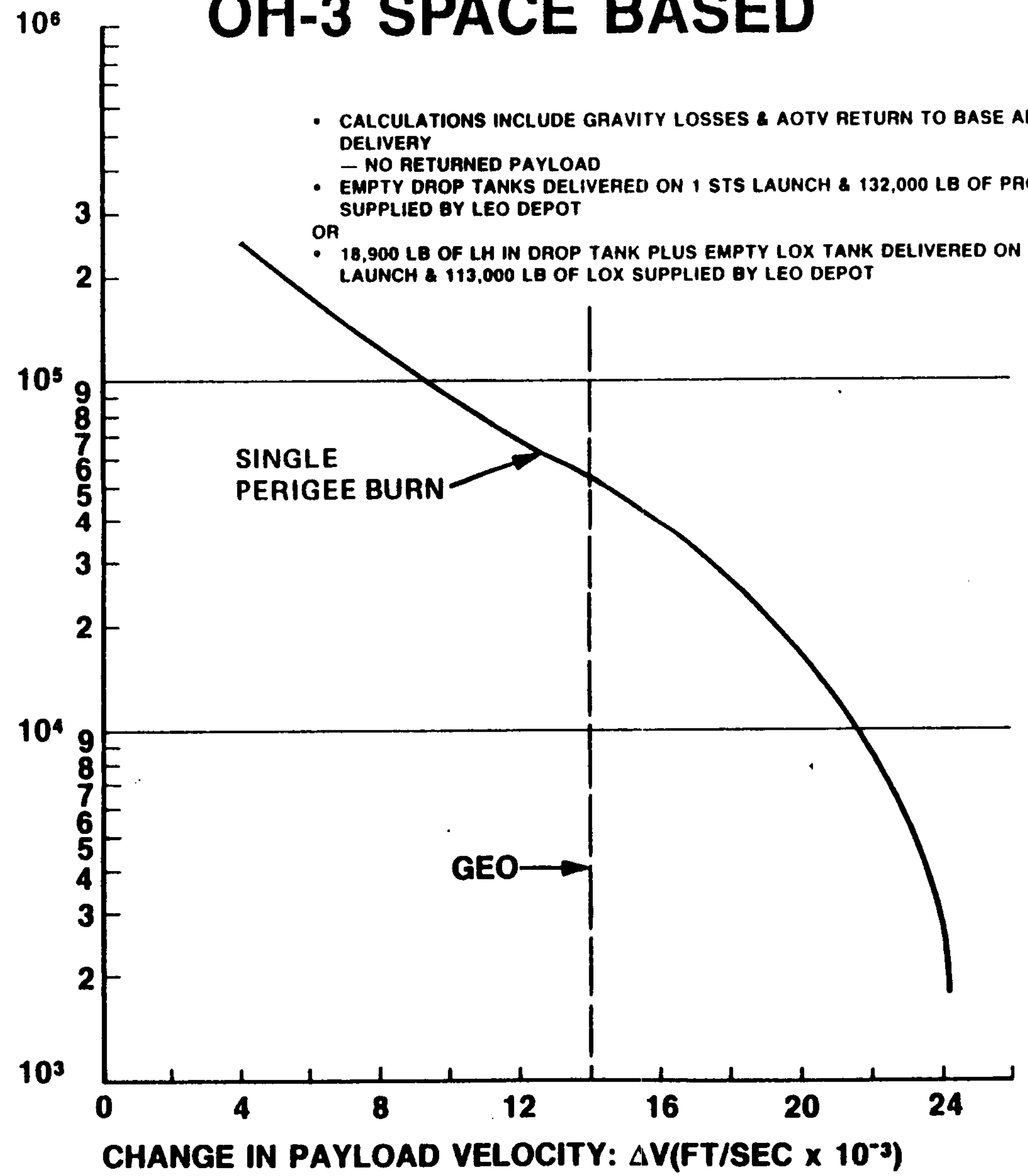
GRUMMAN

V83-2102-005G

The delivery performance of OH-3 with these very large drop tanks has been generalized. The GEO delivery performance previously discussed (55,300 lb to GEO) is shown at the intersection of the vertical line "GEO" and the curve "Single Perigee Burn". The curve allows estimates of the payload delivery capability of the small OH-3 (but with very large tanks) to non-GEO orbits. "Change in Payload Velocity" is the total ΔV needed for the payload to go from its origin to its destination. A combination ΔV, obtained propulsively and aerodynamically for the AOTV, has been included for the return part of the mission.

CAPABILITY FOR HEAVY PAYLOAD DELIVERY: OH-3 SPACE BASED

IRAD



V84-1106-033(T)

The evolution from a payload delivery mission model to a payload delivery/manned service mission model can be accommodated by the approach shown on the facing page.

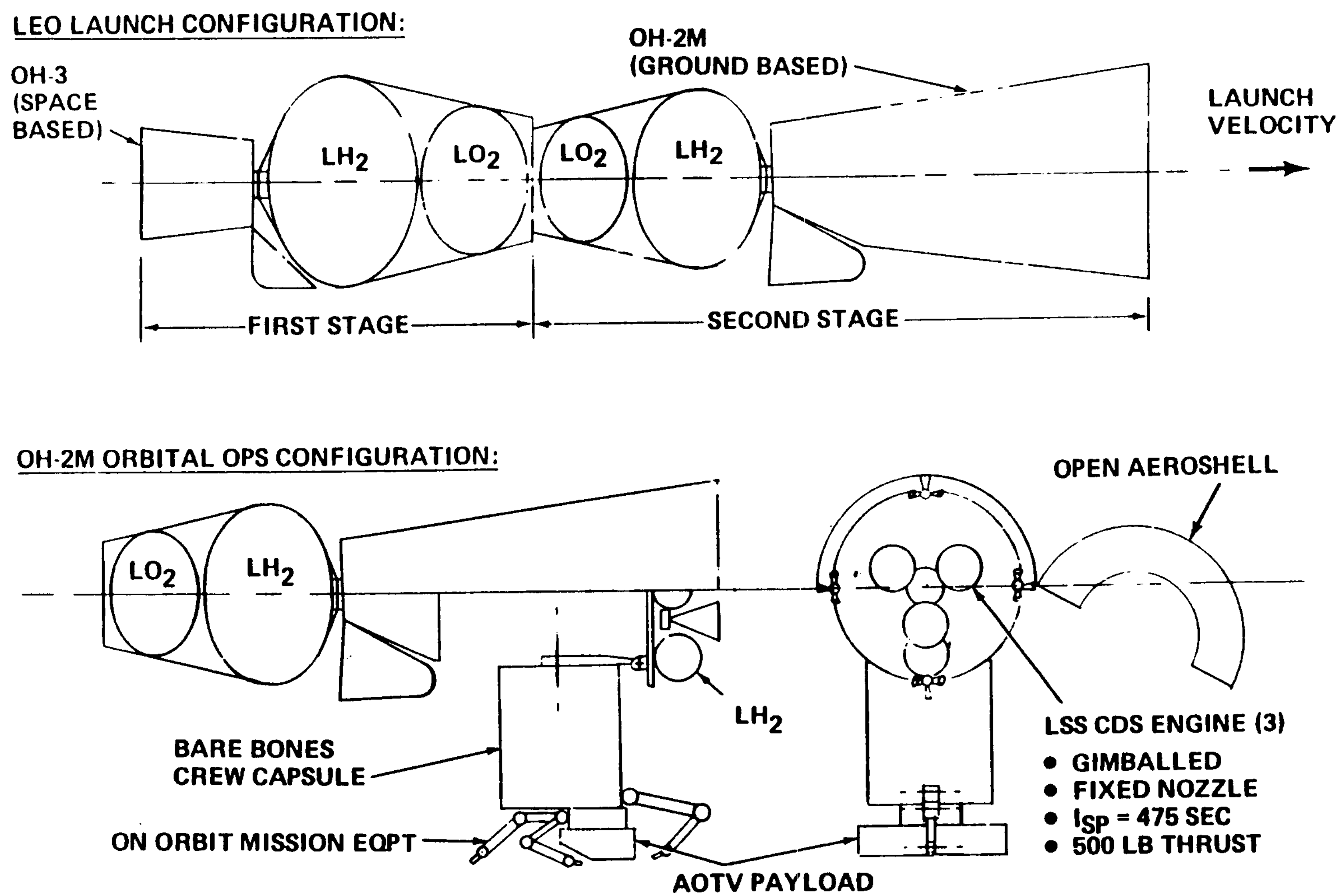
A delivery only vehicle (DOV) such as the OH-3 is used in a two stage configuration with the manned service vehicle OH-2M. The first stage of the system, OH-3 with a set of external tanks, performs the perigee burn. This brings the OH-2M to the apogee of the elliptical transfer orbit. GEO circularization and all subsequent burns at GEO are handled by the OH-2M main propulsion system. both vehicles, OH-3 and OH-2M, release their drop tanks into a burn-up orbit and use their aeroassist capability for rendezvous at LEO. The facing diagrams show the GEO launch configuration and orbital OPS configuration of the OH-3/OH-2M manned service 2 stage vehicle system.

The orbital operations configuration of OH-2M is shown in the lower part of the figure. It uses the same bare bones crew capsule, the same on-orbit equipment, and delivers the same GEO payload (2000 lb) as H-1M. Two features make OH-2M the smallest manned vehicle we have designed in this study. One is small, low thrust engines. Since the main propulsion system only operates against large masses at very high altitudes, low levels of thrust/weight do not incur meaningful gravity losses. The second feature which has produced a small size is that all cryogenic tankage (except for small tanks for post aeromaneuver burns) are external to the aeroshell. This allows a minimum length and diameter AOTV. The size is driven by the geometry of the bare bones crew capsule.

This system of using two vehicles for a manned mission allows significant operational time for a cargo carrying AOTV before the development expense of a man rated vehicle system is incurred. Although total program development costs are higher, the two vehicle system allows the lowest peak annual funding. If the manned vehicle is developed much later than the cargo vehicle, development costs on the cargo vehicle can be recouped before any expenditures for the manned vehicle are initiated.

EVOLUTION OF DOV TO MANNED MISSIONS

IRAD



GRUMMAN

V84-1106-001(T)

The facing figure illustrates the staging breakdown, ΔV requirements, propellant requirements and a performance summary for a bare bones manned mission OH-3/OH-2M vehicle system. Of the two stages, all but the OH-3 propulsion module are ground based. The 63,940 Space Station (SS) release weight contains a 59940 lb ground based component. This consists of the OH-3 drop tanks, OH-2M drop tanks, OH-2M, and the propellant required for the mission. The 4000 lb space based portion of the GLOW is composed of the dry weight of the OH-3.

Note that the significant gravity loss (ΔV = 137 ft/sec) associated with using the modest output of the OH-3 engines (12,000 lb) is based upon a single perigee burn mission. Multiple perigee burns will reduce this and allow more delivered payload at GEO.

Aeromaneuvering is only performed by OH-2M ($\Delta i = 15^{\circ}$). Since OH-3 doesn't perform significant plane changing in its perigee kick role, it only performs aerobraking.

We think the crew capsule shielding needed to protect the crew at GEO will provide adequate crew protection for 2 or 3 passes through the Van Allen Belt radiation field.

OH-2M PERFORMANCE ON MANNED GEO MISSION

IRAD

- SINGLE PERIGEE BURN
- TWO STAGE VEHICLE, 1 1/2 STAGES ARE GROUND LAUNCHED

FIRST STAGE

- OH-3 PROPULSION MODULE (SPACE BASED) WITH ELLIPSOIDAL DROP TANKS

	MASS (LB)
AOTV LAUNCH WEIGHT	4000
PROPELLANT (O/F = 6:1)	27,150
USEABLE (ΔV_1)	(26,250)
USEABLE (ΔV_{1F})	(150)
RESERVE & RESIDUAL	(540)
PREFLIGHT LOSSES	(210)
DROP TANK "1/2 STAGE"	1790
PAYLOAD DELIVERED AT GEO TRANSFER ORBIT	31,000

WEIGHT AT SS RELEASE = 63,940 LB
GLOW = 63,730 LB

ΔV_1 = 8116 FT/SEC
$[T/W]_1$ = 0.184 & GRAV LOSS ≈ 137 FT/SEC

SECOND STAGE

- OH-2M PROPULSION MODULE, "BARE BONES" CREW MODULE, & ELLIPSOIDAL DROP TANKS

	MASS (LB)
AOTV LAUNCH WEIGHT (INCLUDING CREW & EQPT)	11,660
PROPELLANT (O/F = 6:1)	16,300
USEABLE (ΔV_2 & ΔV_3)	(15,420)
USEABLE (ΔV_4)	(270)
RESERVE & RESIDUAL	(330)
INFLIGHT LOSSES	(280)
DROP TANK "1/2 STAGE"	1040
PAYLOAD DELIVERED AT GEO	2000
PAYLOAD RETURNED	0
$GLOW_2$ =	31,000 LB

ΔV_2 = 6000 FT/SEC
ΔV_3 = 5200 FT/SEC, $\Delta i_{AERO} = 15°$
ΔV_4 = 350 FT/SEC

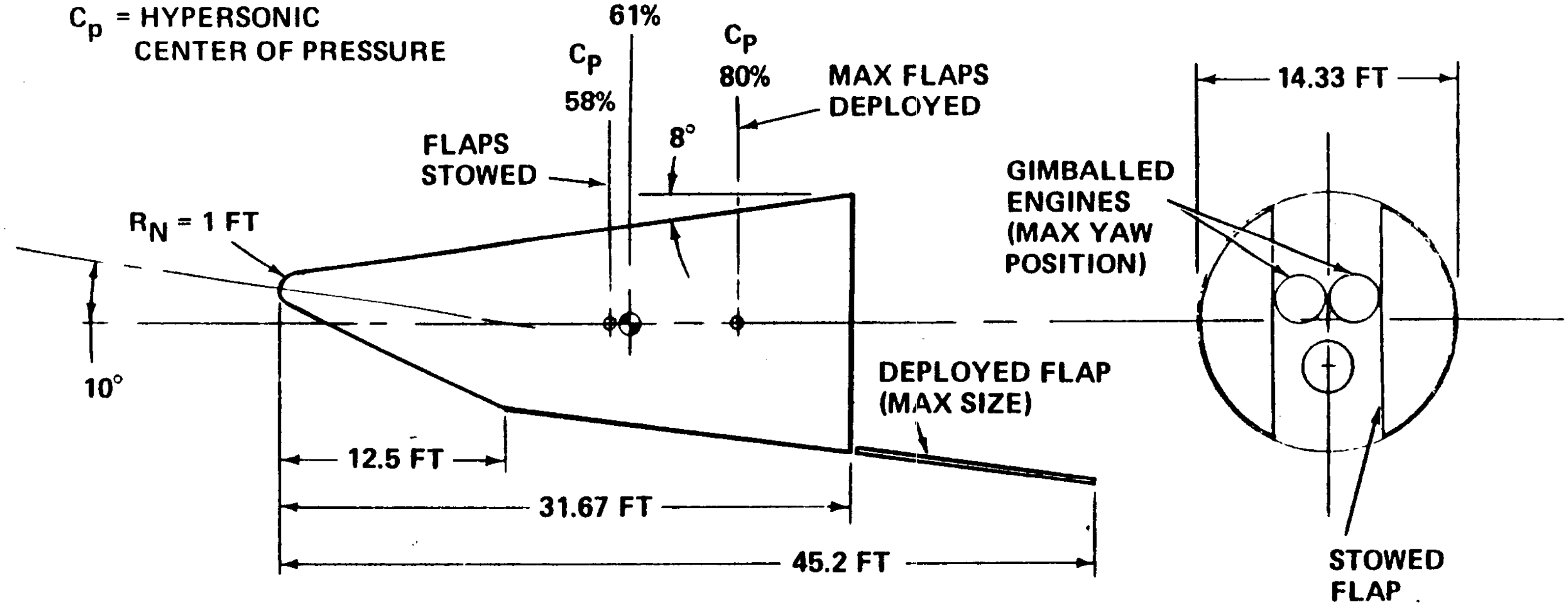
GRUMMAN

V84-1106-007(T)

A drawing of the OH-2M in its re-entry configuration with the expected center of mass is illustrated. The center of pressure locations with the vehicles flaps in both the stowed and deployed positions is also shown. The 3% of body length difference between the "flap stowed C_P" and the estimated center of mass is considered an acceptable spread for this level of design. Obviously a smaller flap size would be acceptable for this configuration.

SMALL MANNED AOTV "OH-2M": ATMOSPHERIC ENTRY

IRAD



$(L/D)_{MAX} = 1.2$
$(\Delta i)_{AERO} = 15°$ FROM GEO

GRUMMAN

V83-2102-008G

The subsystem weight breakdown of the OH-2M is shown.

OH-2M MASS PROPERTIES

IRAD

SUBSYSTEM	MASS	(LB)
CREW CAPSULE		5100
"BARE BONES" CAPSULE (2 MEN)	3580	
ON-ORBIT MISSION EQPT	1320	
DEPLOYMENT	200	
STRUCTURE		2040
SHELL	685	
SUPPORT	545	
FUEL TANK	10	
OXIDIZER TANK	10	
FLAPS	540	
UNBERTHING/LATCHING	250	
THERMAL PROTECTION SYSTEM		390
TANK INSULATION	25	
SHELL INSULATION	365	
PROPULSION		
ENGINE (THREE 500 LB LSS-CDS)	350	1020
GIMBALS & ACTUATORS	100	
PLUMBING	570	
ACS		450
EPS		900
AVIONICS		700
	TOTAL	10,600 LB
	10% CONTINGENCY	1060
	TOTAL DRY WT =	11,660 LB

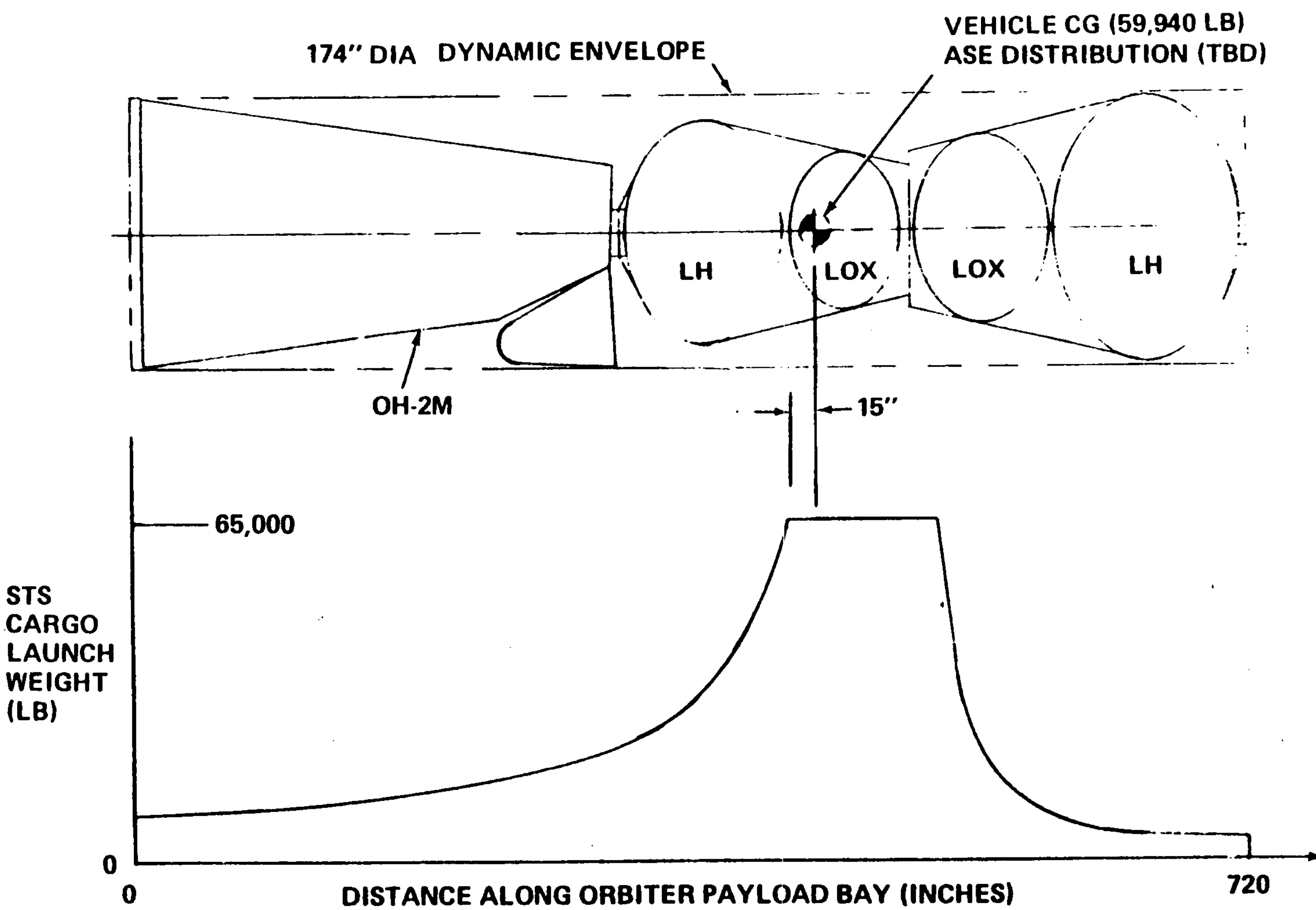
GRUMMAN

V83-1652-062W

The facing figure illustrates the Orbiter cargo bay packaging of the OH-2M manned mission components. The cargo bay carries the OH-2M and the two sets of full drop tanks required for the mission. The 59,940 lb cargo meets the length, weight and center of mass (CG) constraints of a 65,000 lb capacity shuttle.

OH-2M: LAUNCH CONFIGURATION WITHIN ORBITER

IRAD



V84-1106-002(T)

Program costs associated with developing and building the first operational OH-2M are shown. The STS user costs of one space flight test have been included in total program costs of $1.29B.

Since the OH-2M costs are about 7% less than the total program costs for a single stage H-1M ($1.38B, reported in Phase I) there appears to be a small cost advantage to the two-stage approach to manned missions. However, loss of mission flexibility and higher orbital operations costs may quickly offset the estimated savings of $90M.

CONTRACTOR PROGRAM COSTS ('84$): OH-2M

IRAD

- SPARES + FIRST UNIT + DDT&E = $1.29 B

	COST ELEMENT	PROGRAM COSTS IN MILLIONS (1984 $)			
		DDT&E	1ST UNIT	SPARES	OPS
1.1	PROJ MGMT	19.2	3.7		0.08
1.2	SYST ENG & INT	60.1			
1.3	SPACE VEHICLE	806.8	153.5	12.3	4.5
	CREW MODULE	(282.8)	(75.7)	(6.2)	
	VEHICLE MODULE	(505.2)	(73.5)	(6.0)	(0.1)
	DROP TANKS	(18.8)	(4.4)	(0.4)	(4.4)
1.4	GRND SUPPORT SYST	103.8			
1.5	MISSIONS OPS	5.6			3.9
1.6	FLT SUPPORT EQ (ASE)	90.7			
1.7	SPACE TRANSPORT*				84.1*
1.8	INSTL, ASSY, & C/O		29.6		
1.9	SYST TEST & EVAL	18.4			
	PROGRAM TOTALS	$1104.7M	$187.3M	$12.5M	$92.6M

*"SPACE TRANSPORT" IS NASA CHARGE TO A USER OF STS CARGO BAY

- NO FLIGHT TEST SINCE AEROMANEUVERING TECHNOLOGY HAS BEEN ESTABLISHED BY SMALL DELIVERY ONLY AOTV (OH-3)

GRUMMAN

V83-1652-058V

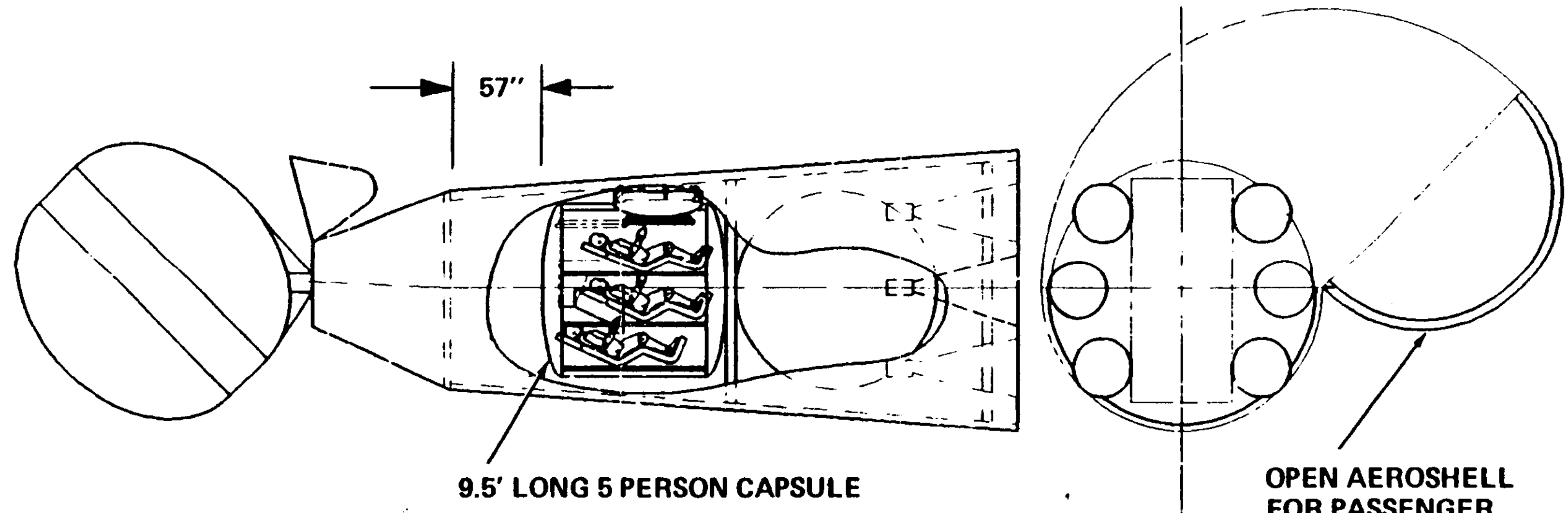
A way in which the H-1M AOTV may be transformed from a satellite servicing vehicle, with a crew of 2, to an austere transport vehicle for 5 people is illustrated. The Bare Bones capsule has been replaced by a 9-1/2 foot long 5 person capsule. This leaves nearly a 5 foot opening inside H-1M, which cold accept 3600 pounds of cargo. The couch angles within the 5 person capsule have been established so that the resultant of aerodynamic forces (at L/D = 1.0) is perpendicular to the backs of the passengers. People can readily accommodate 3 times gravity (3 "g") accelerations in this position.

Like H-1M, this system can perform a round trip to GEO mission from a single STS launch (with a 65K capacity Orbiter). The 5 passengers could travel from Earth to LEO in the Orbiter cabin. They could transfer to the 5 person cabin via IVA tunnel or docking port, after the AOTV has been prepared for its mission.

H-1M: 5 PERSON TRANSPORT TO GEO

IRAD

- SAME AOTV AS H-1M
  - REMOVE "BARE BONES" 2 MAN CAPSULE
  - ADD 5 PERSON TRANSPORT CAPSULE & CARGO
    - UP TO 3600 LB OF CARGO



9.5' LONG 5 PERSON CAPSULE

- COUCH ANGLE SET BY $\alpha = 20°$ & L/D = 1
- CONTROL STATION PERMITS ON-BOARD PILOT
- VOLUME BETWEEN AOTV & CAPSULE PERMITS MOVEMENT OF CENTER OF MASS &/OR CARGO TRANSPORT

OPEN AEROSHELL FOR PASSENGER INGRESS/EGRESS

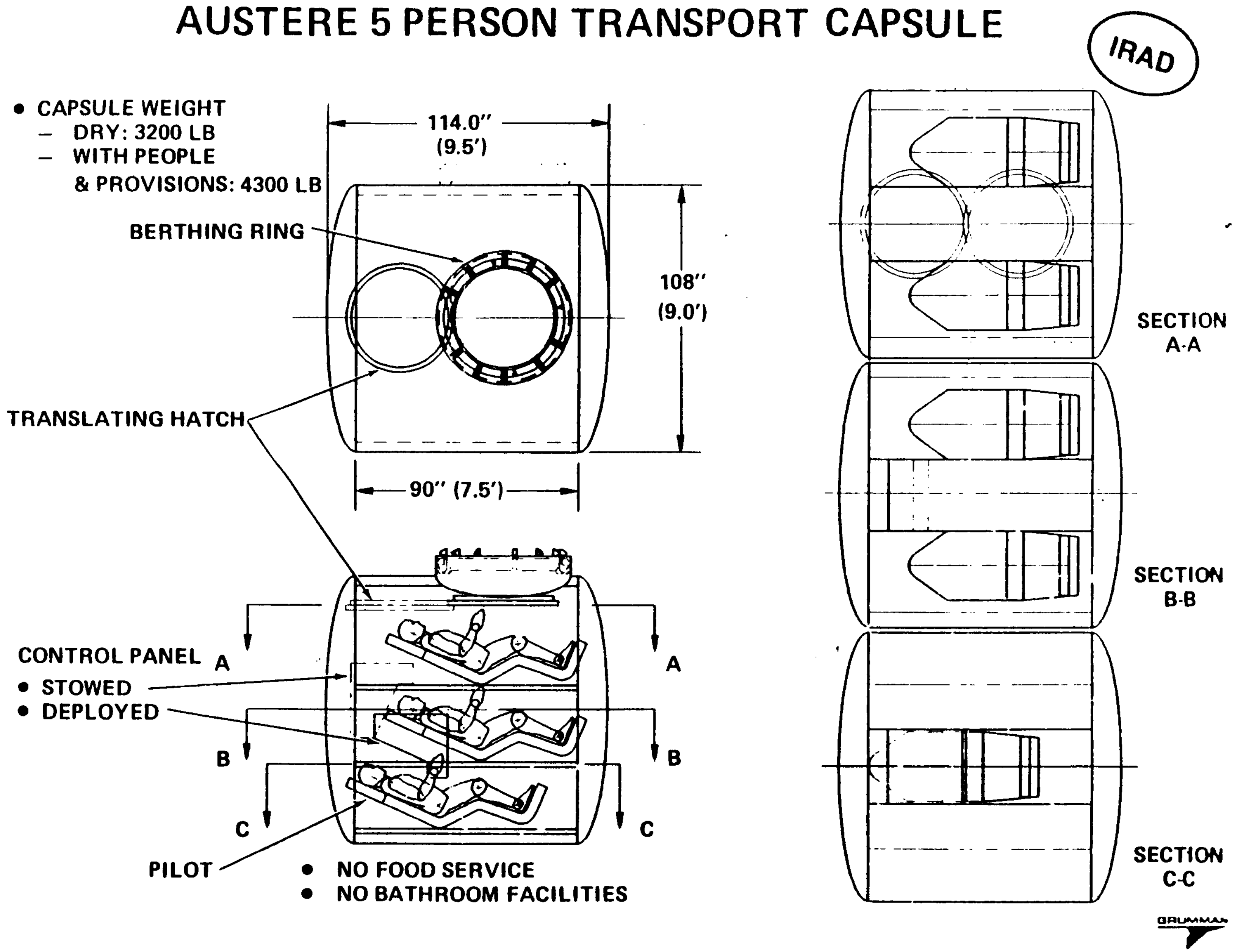
GRUMMAN

V84-1106-003(T)

The general arrangement of an austere 5 person transport capsule is shown. Since Hohmann Transfer transit time from LEO to GEO is on the order of 5 hours, no provisions have been made within the capsule for bathroom facilities or large quantities of food. The limited volume requires that most persons remain in their couches for the duration of the mission, although some volume remains open between the two stacks of couches for 1 or 2 persons to move their bodies into different positions.

Sections A-A, B-B and C-C are all shown as if the sections were taken at B-B, where the full diameter of the cylindrical portion of the capsule occurs. However, in locating the couches within the capsule, consideration has been given to the narrower sections at A-A and C-C when trying to fit the couch arrangements within the cylindrical body. Sufficient clearance exists at these locations for the arrangements.

AUSTERE 5 PERSON TRANSPORT CAPSULE



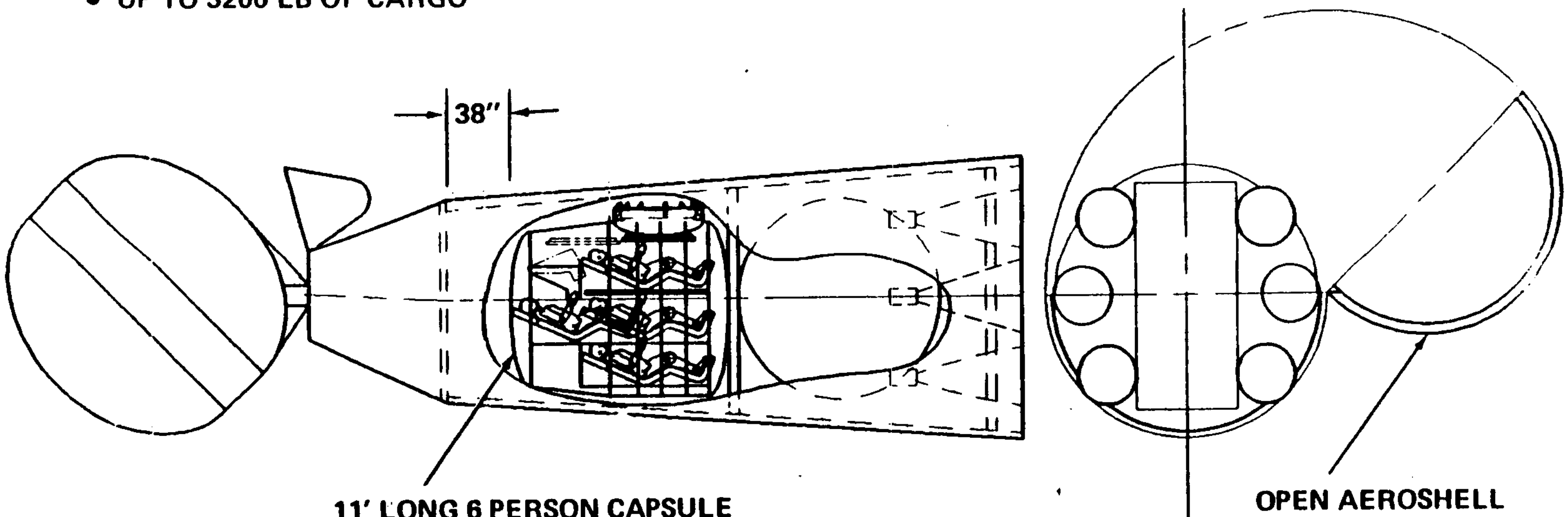
V84-1106-004(T)

The installation of a 6 person transport capsule within H-1M is shown. This larger capsule leaves 3 feet of open space within H-1M for the addition of 3200 pounds of cargo while delivering 6 persons to GEO.

H-1M: 6 PERSON TRANSPORT TO GEO

IRAD

- SAME AOTV AS H-1M
  - REMOVE "BARE BONES" 2 MAN CAPSULE
  - ADD 6 PERSON TRANSPORT CAPSULE & CARGO
    - UP TO 3200 LB OF CARGO



11' LONG 6 PERSON CAPSULE

- COUCH ANGLE SET BY $\alpha = 20^\circ$ & L/D = 1
- CONTROL STATION PERMITS ON BOARD PILOT
- VOLUME BETWEEN AOTV & CAPSULE PERMITS MOVEMENT OF MASS CENTER &/OR CARGO TRANSPORT

OPEN AEROSHELL FOR PASSENGER INGRESS/EGRESS

GRUMMAN

V84-1106-005(T)

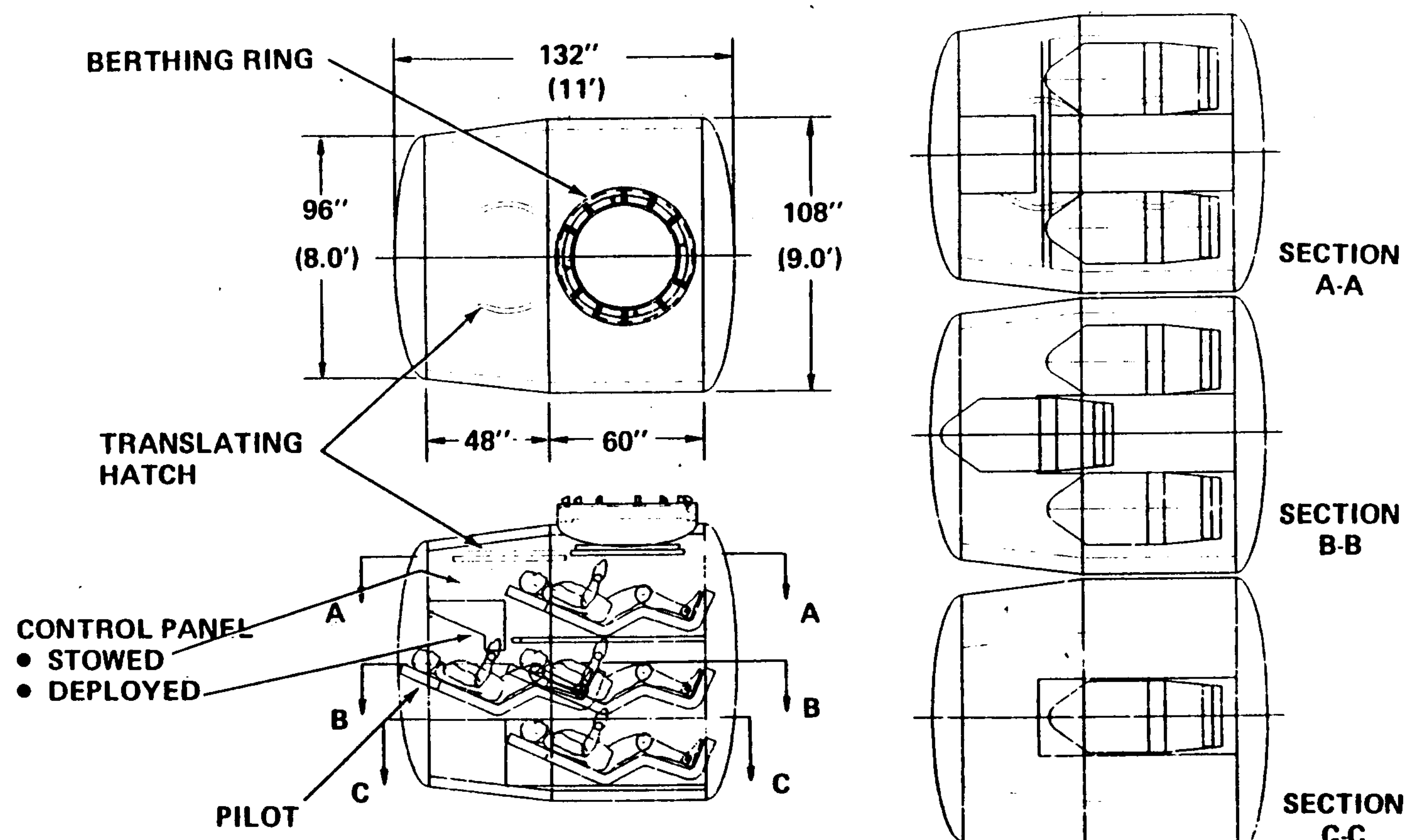
The general arrangement of an austere 6 person transport capsule is shown. In this configuration, the pilot has been located at the forward region of the middle level. Although the AOTV could be flown automatically or from some remote location, we think providing on board piloting capability is a prudent measure for manned missions.

Sections A-A, B-B and C-C are all shown as if the sections were taken at B-B, where the full diameter of the cylindrical portion of the capsule occurs. However, in locating the couches within the capsule, consideration has been given to the narrower sections at A-A and C-C when trying to fit the couch arrangements within the cylindrical body. Sufficient clearance exists at these locations for the arrangements shown.

AUSTERE 6 PERSON TRANSPORT CAPSULE

IRAD

- CAPSULE WEIGHT
  - DRY: 3365 LB
  - WITH PEOPLE & PROVISIONS: 4725 LB



- NO FOOD SERVICE
- NO BATHROOM FACILITIES

GRUMMAN

V84-1106-006(T)

A weight breakdown for the 5 and 6 person austere transport capsules is shown. The 420 pound difference between the two capsules results from the additional person, and his supplies, and the enlarged structure necessary to enclose a larger volume. These capsule weights are slightly less (6%) than the Bare Bones capsule weights reported in Phase I of this study.

AUSTERE PEOPLE TRANSPORT
CAPSULE WEIGHTS

	WEIGHT-LB 5 MAN	6 MAN
STRUCTURE & MECHANISM		
SHELL	1215	1340
SECONDARY		
BERTHING RING	220	220
HATCH	15	15
HATCH TRACK	25	25
STAB & CONT	38	38
NAV & GUIDANCE	76	76
CREW PROVISIONS		
COUCHES (3G)	125	150
MISC	150	150
INSTRUMENTATION	130	130
ECLS	300	325
EPS	300	300
COMMUNICATION	60	60
CONTROLS & DISPLAYS	230	230
CONTINGENCY (10%)	288	306
TOTAL DRY WT	3172	3365
CONSUMABLES (BASED ON 48 HR)	(1134)	(1360)
FOOD 5 lb/MAN DAY	55	66
H_2O 5.2 lb/MAN DAY DRINKING	52	63
O_2 1.84 lb/MAN DAY + LEAKAGE	20	24
N_2	8	8
LIOH 5 lb/MAN DAY + CANISTERS	61	73
CREW 170 lb	850	1020
CREW CABIN SUITS (18 lb/SUIT)	88	106
TOTAL WT	4306 LB	4725 LB

IRAD

GRUMMAN

V84-1106-008(T)

The preceeding set of subjects on evolutionary capabilities of biconic AOTVs which were designed for specific missions are summarized. The major message is that modularity allows for substantial increases or changes in mission capability. Also, that AOTVs do not have to be used in an aeroassisted mode for infrequent missions that require tanks external to the biconic aeroshell. For these missions, all propulsive operation of an AOTV allows reuse of the external tanks.

SUMMARY OF EVOLUTIONARY CAPABILITIES

- HEAVY (50K) PAYLOADS CAN BE DELIVERED TO GEO USING SMALL AOTV
  - OH-3 USED IN SINGLE STAGE MODE
  - IF TANK RECOVERY IS IMPORTANT, USE OH-3 IN ALL PROPULSIVE MODE AND ACCEPT LESS DELIVERED PAYLOAD ON THESE INFREQUENT MISSIONS
- TWO STAGE MANNED GEO MISSION POSSIBLE ON SINGLE 65K STS WITH SPACE BASED OH-3
  - OH-2M PAYLOAD = H-IM PAYLOAD
  - EXTERNAL O_2 AND H_2 TANKS GIVE OH-2M THE SAME MISSION FLEXIBILITY OF OH-3
- 5 OR 6 PERSON (PAYLOAD) TRANSPORT TO GEO POSSIBLE WITH H-IM OR OH-2M
  - REMOVE "BARE BONES" CAPSULE & ADD TRANSPORT CAPSULE.

GRUMMAN

V84-1106-026(T)

3.2.2.4 New Modularized AOTV Configurations for Both Cryogenic and Storable Propellants

The vehicles in the next group of figures represents a departure from the vehicles designed earlier in this contract. The previously designed vehicles are dedicated to a particular mission. Generally, a vehicle had been either a payload delivery vehicle or a manned GEO service vehicle. Little mission flexibility is available with this type of AOTV design if there is only 1 AOTV. A second design approach is outlined below.

The vehicles listed on the bottom half of the figure employ modular block construction. This modularity has several advantages, the two greatest being tailoring of the vehicle to the required mission is possible and fewer access problems are experienced during space service operations. The vehicle is prepared for a mission by decoupling or assembly of the required modular components.

All subsystems have been designed to our selected man-rating criteria, fail safe/fail safe. However, when this level of redundancy is not required by a cargo vehicle, it is contained in the manned module, where possible. Propulsion system redundance is in the propulsion module. Therefore there is no minimal additional hardware weight penalty added for redundancy considerations in the cargo modules.

Three missions were used for sizing modular vehicles:

A) A Grumman/GE recommended 8K delivery, 6K return manned service mission to GEO.

B) The NASA 14K delivery 14K return manned service mission to GEO.

C) A 13K payload delivery to GEO with no return payload.

Each of the vehicles is designed with internal tankage sized for one of the three missions. All manned missions are accomplished by 1 or 1-1/2 stage delivery single perigee burn Hohmann transfer modes. Payload delivery missions use perigee kick delivery:

- o Single perigee burns
- o Apogee burn provided by payload
- o Payload weight and size increased to adjust for apogee burn.

The table classifies the AOTV design based on two parameters. The first parameter is the sizing of the internal tankage. The internal tankage of the vehicle is sized to one of the three previously discussed missions. Other missions with differing propellant requirements are accomplished by off-loading of propellant tanks (in the case of missions with lower propellant requirements) or by the addition of auxiliary drop tanks (in the case where propellant requirement exceeds the volume of the internal tanks). The second classification parameter is the date at which the required technology for the AOTV is expected to be available.

Generally, the required AOTV technology is driven by the required engine technology. Three timeframes for the OTV designs were investigated. A storable propellant (N_2O_4/MMH) engine technology will be available in 1987 (expendable use). It represents a relatively high performance storable engine available in the near term. A study for an advanced LO_2/LH_2 engine technology is being overseen by LeRC. LeRC projects that the required advanced cryogenic engine technology will be available in 1995. The third timeframe is for an advanced storable propellant engine using the propellant combination N_2F_2/N_2H_4. The engine technology dates and engine availability dates are TBD.

* The vehicle identification code used in this report is described below:

$2 - N_2O_4$ /MMH

Oxidizer Fuel

series design

number for propellant combination (followed by M i manned vehicle)

The flow path of arrows in the diagram illustrates the order in which the vehicles are presented in this report.

NEW AOTV DESIGNS

- MODULAR AOTV DESIGNS THAT UTILIZE PARTS OF THE SAME VEHICLE FOR UNMANNED & MANNED MISSIONS
  - A = INTERNAL TANKAGE SIZED FOR 8K UP & 6K BACK MANNED GEO MISSION
  - B = INTERNAL TANKAGE SIZED FOR 14K UP & 14K BACK MANNED GEO MISSION
  - C = INTERNAL TANKAGE SIZED FOR GEO PAYLOAD DELIVERY OF 12K TO 13K
- AOTV DESIGNS, IN ORDER OF APPEARANCE

A	B	C	REQ'D TECHNOLOGY AVAILABLE BY
2 – N_2O_4/MMH			1987
		3 – N_2O_4/MMH	1987
	4M – LO_2/LH_2		1995
1 – N_2F_4/N_2H_4			TBD
		3 – N_2F_4/N_2H_4	TBD

GRUMMAN

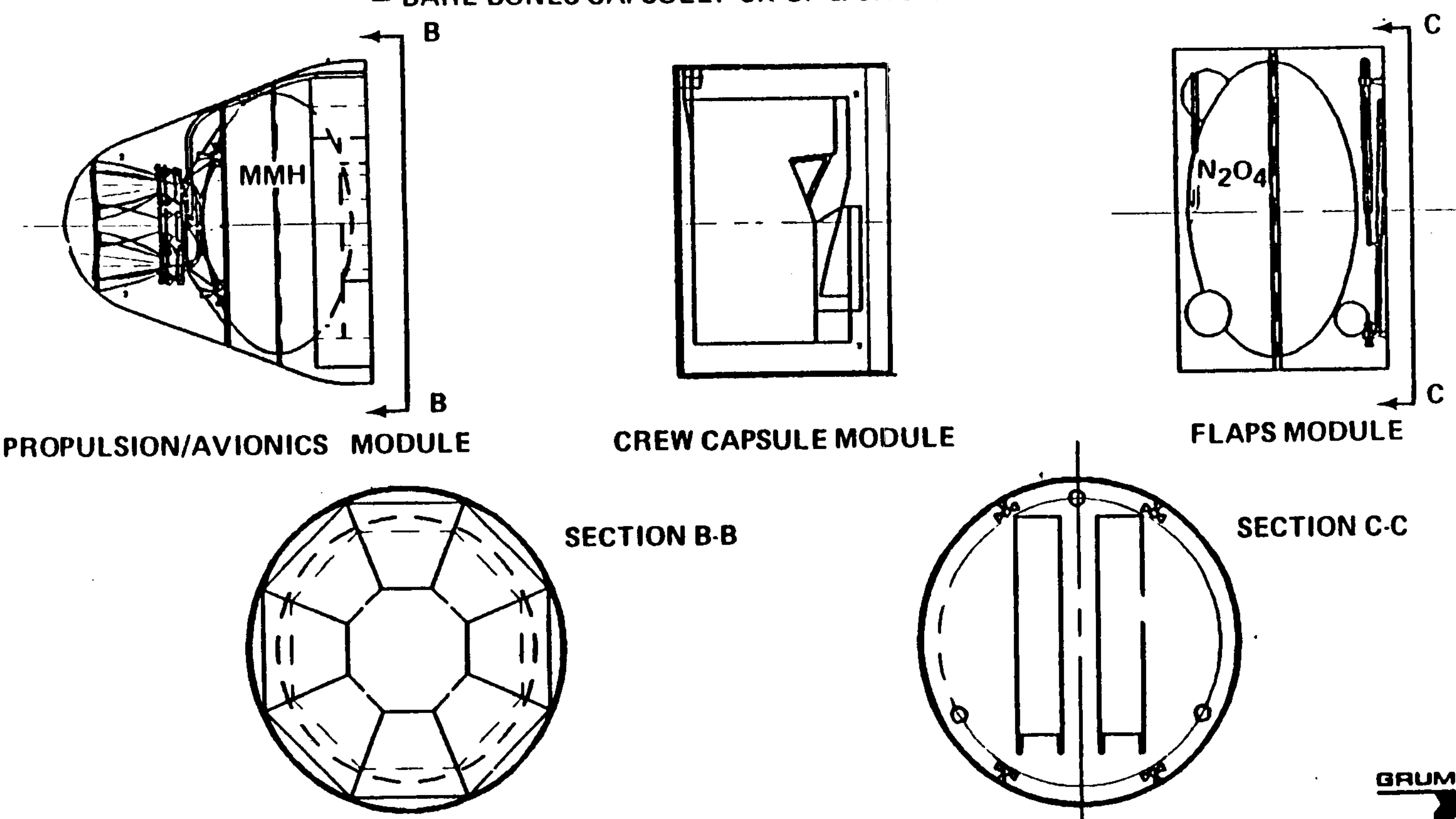
1590-009(T)

The 2-N_2O_4/MMH is a near term technology storable vehicle with internal tankage sized for an 8K delivery, 6K return, manned GEO service mission. Shown are the modular pieces used to assemble 2-N_2O_4/MMH series vehicles. The flaps module, crew capsule module, and propulsion avionics module are all that are required to assemble either a delivery vehicle or a manned service vehicle.

The flaps module contains the oxidizer tank, ACS propellant tanks, ACS thruster packs, aeroassist equipment and the payload ring pickup points. The crew capsule module is a cylindrical slug and contains the crew capsule and related deployment mechanisms. The propulsion avionics ring contains the fuel tank, avionics ring, ACS thrusters and 3, 3750 lbf, I_{SP} = 343 lbf sec/lbm N_2O_4/MMH storable engines. Each interface between modules must contain connect/disconnect fittings for electrical, fluid and structural connections. The following two figures illustrate the 2-N_2O_4/MMH in its delivery configuration and manned service configuration.

2 – N_2O_4/MMH MODULAR COMPONENTS

- 1987 TECHNOLOGY AOTV
- THREE ENGINES TOTAL THRUST = 11250 lbf, I_{SP} = 343 lbf sec/lbm
- VEHICLE TANKAGE SIZED FOR MANNED SERVICE TO GEO
  – BARE BONES CAPSULE: 8K UP & 6K BACK



GRUMMAN

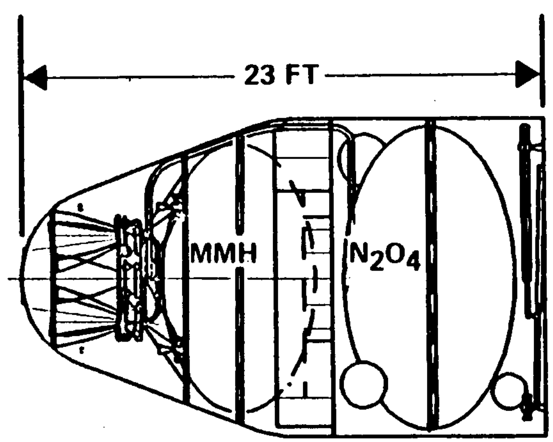
1590-010(T)

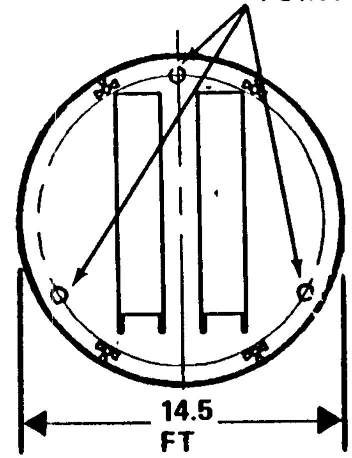
The 2-N_2O_4/MMH as a delivery vehicle is illustrated. The propulsion avionics module has been coupled with the flaps module to create a large capacity vehicle. The vehicle's nose deploys open and the engines are gimbaled outward for engine firing.

When used in a perigee kick mode, the vehicle is capable of delivering approximately 34,000 lbs of usable payload to GEO (after eliminating mass associated with apogee propulsion). This is 5-6 times the average GEO payloads manifested for 1995 or 2000 on one OTV flight, and may be more payload than is logistically practical in the near term. The 159,000 lb GLOW of this system is far beyond the capacity of the STS, and implies that this vehicle must suffer the performance losses of severe propellant off-loading when used as a ground based shuttle compatible AOTV. However, the vehicle is shuttle transportable. If large payload manifests and space basing of OTV's is the selected operating scenario, then 2-N_2O_4/MMH is a viable vehicle concept.

2 – N_2O_4/MMH DELIVERY VEHICLE

DRY WEIGHT	7000 lb
PROPELLANT CAPACITY	90,000 lb
GLOW	159,000 lb
PERIGEE KICK DELIVERY MODE	
• PAYLOAD TO TRANSFER ORBIT	62,000 lb
• USEABLE PAYLOAD AT GEO	34,000 lb





GRUMMAN

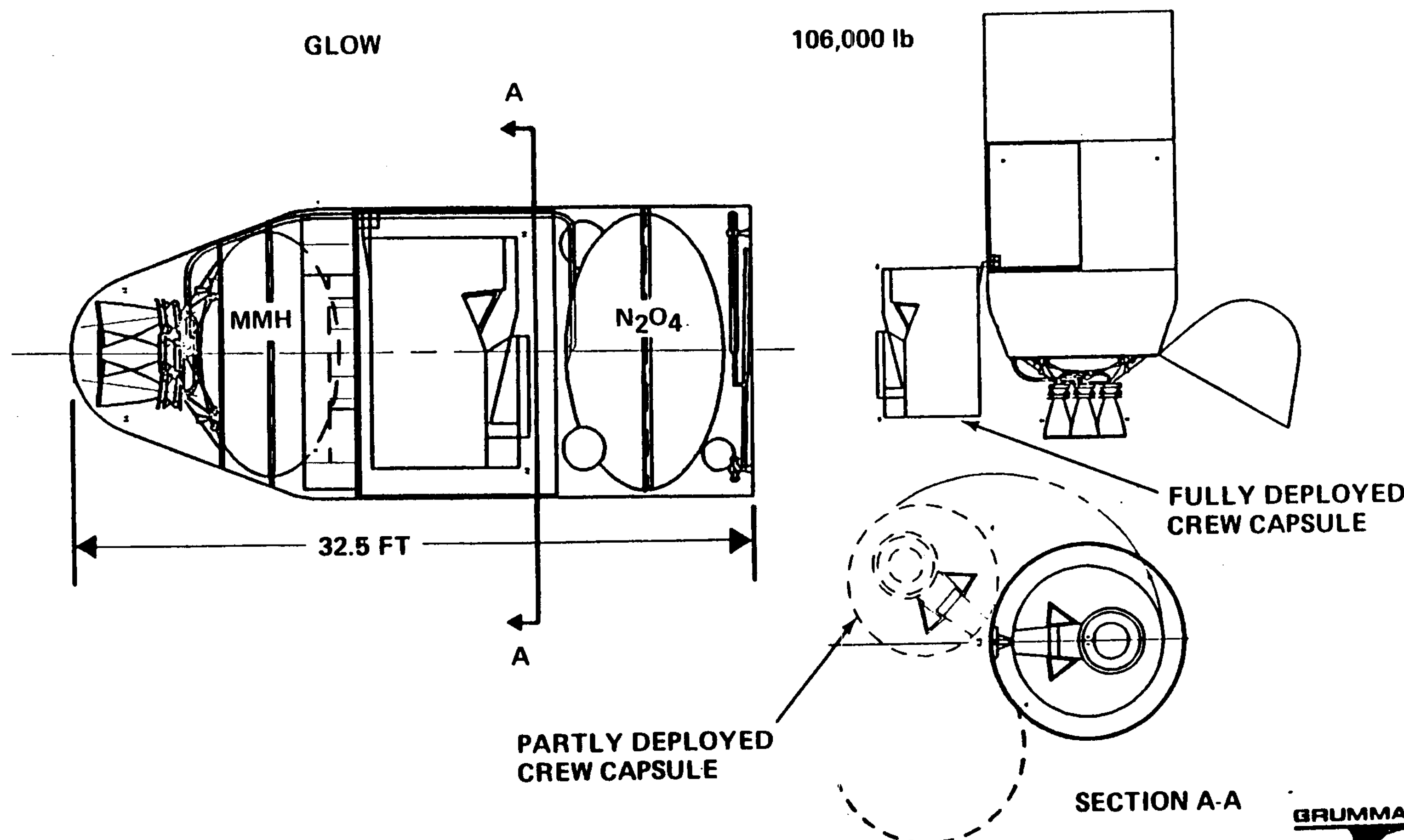
1590-011(T)

The 2M-N_2O_4/MMH is the manned version of the 2-N_2O_4/MMH. The vehicle is created by decoupling the flaps module from the propulsion/avionics module (in the delivery vehicle mode) and inserting a cylindrical slug between the two modules. This slug contains the crew capsule, which is deployed in a three stage process. First the aeroshell door is rotated open. This is followed by a similar rotation of the crew capsule out of the aeroshell. The final position of the crew capsule is obtained by a rotation orthogonal to the first two rotations which brings the capsule to its fully deployed position as shown in the figure. This position allows the crew free access to a worksite.

The vehicle, although not single STS ground launch compatible, is shuttle transportable and can perform an 8K up 6K return manned GEO service mission with a 90K lb propellant load.

2M – N_2O_4/MMH MANNED VEHICLE

DRY WEIGHT	13,900 lb
PAYLOAD LEFT AT GEO	2,100 lb
PROPELLANT CAPACITY	90,000 lb
GLOW	106,000 lb



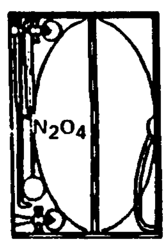
GRUMMAN

1590-012(T)

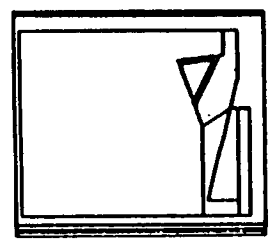
The 3-N_2O_4/MMH vehicle follows very closely the design approach taken on 2-N_2O_4/MMH, with the exception that the internal tankage is now sized for a 12,000 lb usable payload delivery to GEO. Auxiliary external tankage is used to increase the propellant capacity to the level required for an 8K lb up 6K return manned GEO mission or a higher weight GEO cargo mission. The following two figures illustrate 3-N_2O_4/MMH in the payload delivery configuration and manned GEO service mission configuration.

3 – N_2O_4/MMH MODULAR COMPONENTS

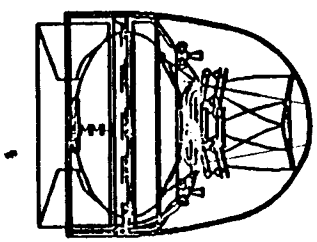
- 1987 TECHNOLOGY AOTV
- THREE ENGINES TOTAL THRUST = 11250 lbf, I_{SP} = 343 lbf sec/lbm
- INTERNAL TANKAGE SIZED FOR 12,000 lb OF USABLE PAYLOAD TO GEO IN PERIGEE KICK MODE



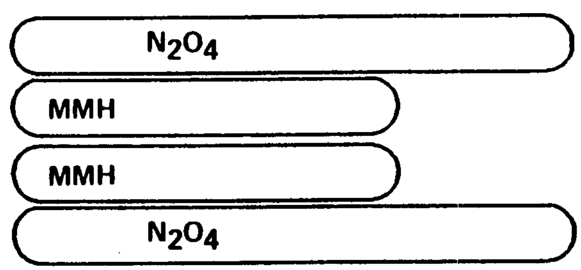
FLAPS MODULE



CREW CAPSULE MODULE



PROPULSION AVIONICS MODULE



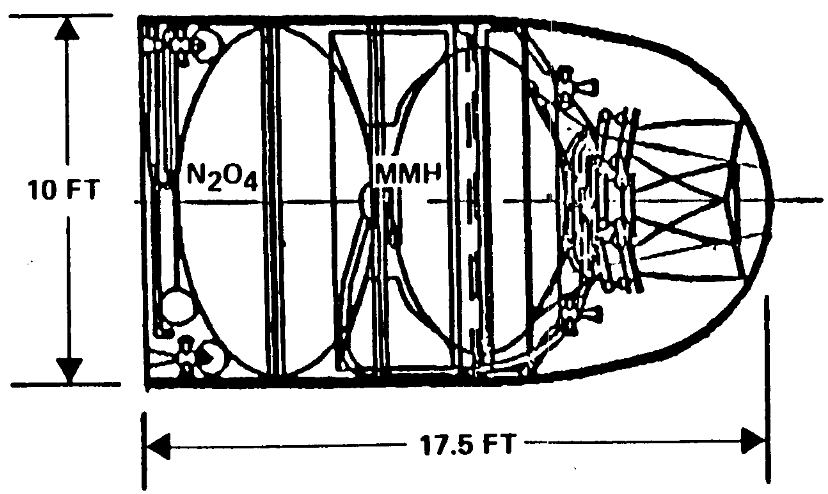
GRUMMAN

1590-013(T)

The 3-N_2O_4/MMH is shown in the figure as a delivery vehicle. The combination of the flaps module with the propulsion/avionics module yields a vehicle that will deliver 12,000 lb of usable payload to GEO. The 60,600 lb GLOW implies that the vehicle is capable of a fully tanked payload delivery mission within the confines of a single 65K shuttle flight. The vehicle configuration allows the payloads to be manifested to the pickup points on the end of the flaps module so that the payloads may be manifested to the vehicle on the ground. Only a deployment of the vehicle nose (to an open position) for engine firing is required once the upper stage package (payload and AOTV) leaves the cargo bay. This configuration is particularly attractive because, as a near term technology vehicle (1987 engine technology), the 3-N_2O_4/MMH could start out as a ground based vehicle and mature into a space based vehicle as space station facilities become available for payload manifesting, system checkout, OTV storage and OTV maintenance.

3 – N_2O_4/MMH DELIVERY VEHICLE

DRY WEIGHT	5300 lb
PROPELLANT CAPACITY	33,000 lb
GLOW	60,600 lb
PERIGEE KICK DELIVERY MODE	
• PAYLOAD TO TRANSFER ORBIT	22,300 lb
• USEABLE PAYLOAD TO GEO	12,000 lb



GRUMMAN

1590-014(T)

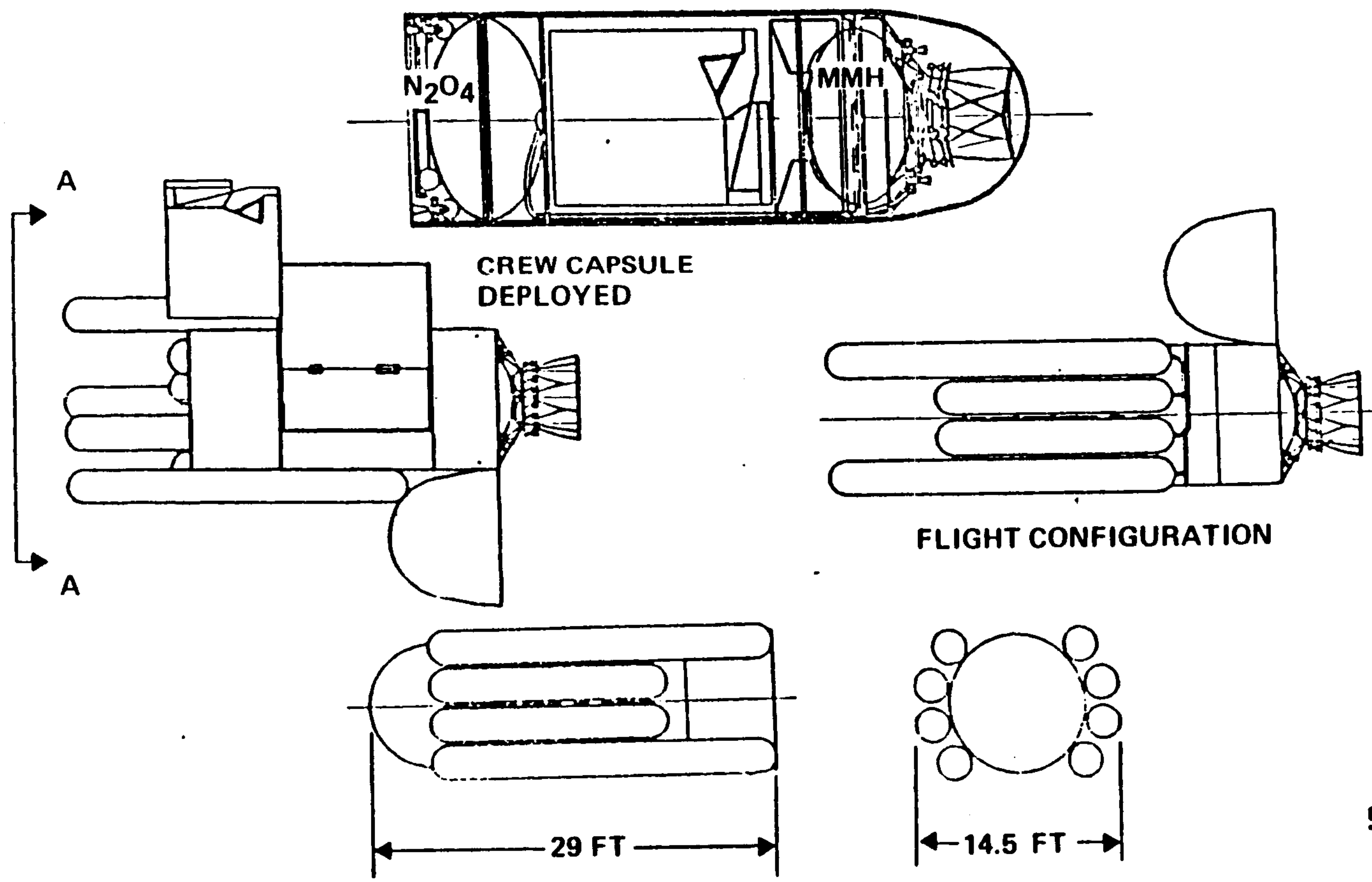
In the growth of 3-N_2O_4/MMH from a payload delivery vehicle to a manned service vehicle, two modules must be added to the system. A crew capsule is placed into the center of the delivery stage. Auxiliary tankage required for the manned mission is added to the crew capsule module. The auxiliary tankage has two sets of pickup points, one for shuttle transport, and the second for AOTV flight. It is sized so that the additional propellant quantity allows the vehicle to perform the 8K up 6K return manned GEO service mission. The auxiliary tanks are thrown into a burn-up orbit prior to AOTV entry into the atmosphere - the aeroassist maneuver.

The crew capsule is deployed in a three state process. The aeroshell door, with auxiliary tanks attached, is rotated open. This step is followed by a similar rotation of the crew capsule out of the vehicle. Finally, a rotation of 90^o about an orthogonal axis brings the capsule to its servicing position. All propellant in the external tankage has been consumed before rotation of the aeroshell door occurs.

The 3M-N_2O_4/MMH, although not single shuttle launch/OTV launch compatible, is shuttle transportable and will allow up to 31' of the cargo bay space for other user payloads.

3M – N_2O_4/MMH MANNED VEHICLE

DRY WEIGHT	13,700 lb
PAYLOAD LEFT AT GEO	1800 lb
DROP TANK/WEIGHT & PLUMBING	900 lb
PROPELLANT CAPACITY	90,000 lb
MAIN TANKS	33,000 lb
AUXILLIARY TANKS	57,000 lb
GLOW	106,400 lb



GRUMMAN

1590-015(T)

A second approach to the development of a modular vehicle is investigated in the design of 4-LO_2/LH_2. Rather than sectioning a cylindrical slug into the center of the vehicle to allow for the addition of a crew capsule, it is possible to add a crew capsule to one end of the vehicle. The design of 4-LO_2/LH_2 provides a base from which evaluations on the relative advantages and disadvantages of this approach to modularity can be evaluated.

The 4-LO_2/LH_2 is a six hinged engine delivery vehicle using 3000 lbf thrust advanced LO_2/LH_2 engine concept with a technology availability date of 1995. The vehicle tankage is sized for a 14000 lb delivery and return GEO manned servicing mission. The single tank has an internal bulkhead and stores both the LO_2 and the LH_2. The LO_2 tank external end cap is a hemisphere. The external end cap on the LH_2 tank is a $\sqrt{2}$ ellipsoid.

The engines are run at a mixture ratio of 7/1 and have a two piece deployable nozzle with an expansion ratio (ϵ) of 1000/1. The engines have a one axis gimbal (hinged) and have an expected specific impulse of 480 lbf sec/lbm.

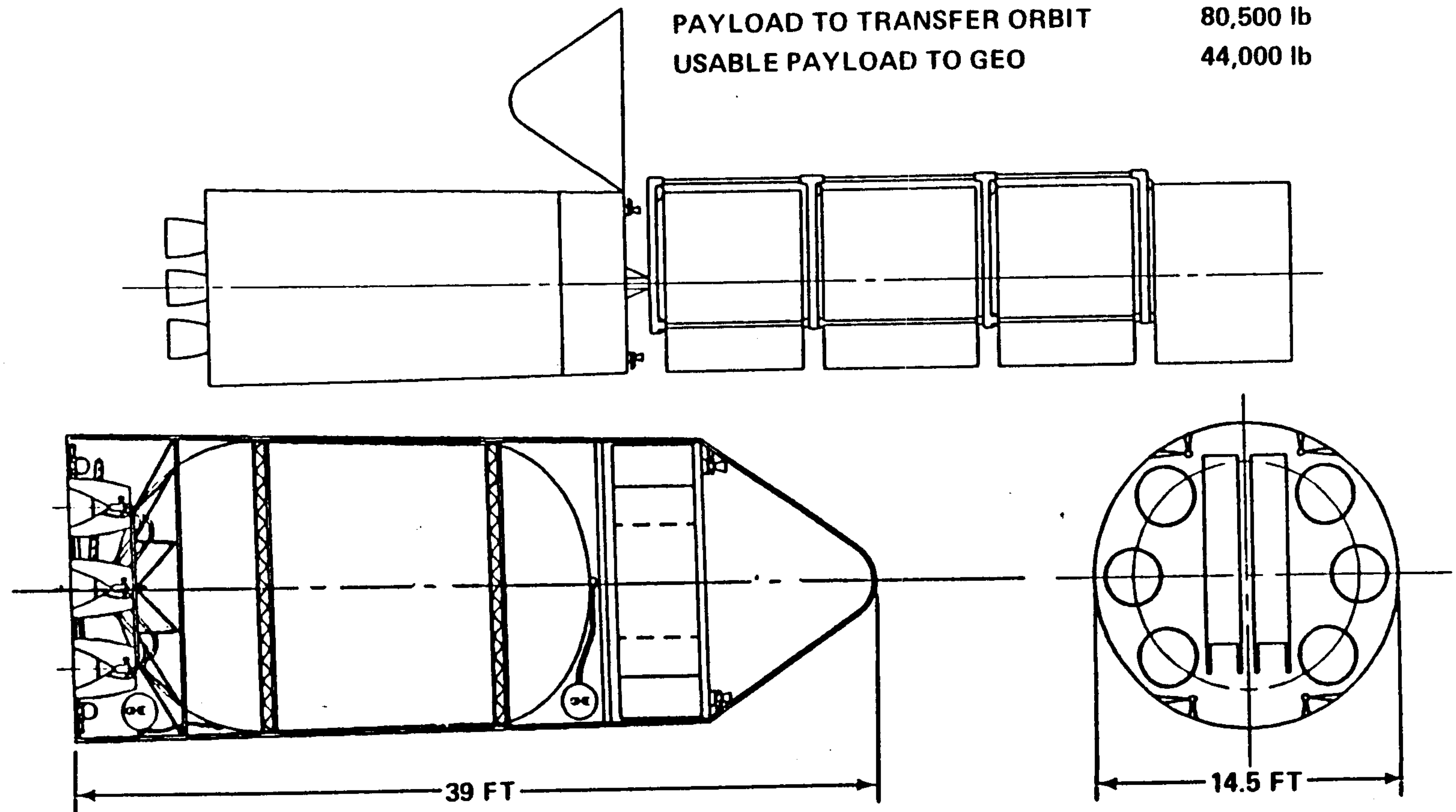
In the interest of serviceability, were possible, all primary failure contributors have been located at module interfaces. ACS thruster packages, propellants and pressurants are in a module form so that they can be changed out as a module without breaking a pressurized line.

The 4-LO_2/LH_2 is shown in its payload delivery configuration. The vehicle is capable of delivering 6 or 7 average 1995-2000 payloads when fully loaded to a 152,500 GLOW. Six to seven payloads is beyond the expected manifesting practice for 1995-2000. Also, the 152,500 GLOW excludes the vehicle from efficient ground based shuttle compatible missions. Severe off-loading would be required for a shuttle compatible launch as a ground based AOTV. This is an inefficient mode of use for a vehicle such as a 4-LO_2/LH_2, whose best function is as a space based AOTV.

4 – LO/LH_2/DELIVERY VEHICLE

- 1995 TECHNOLOGY AOTV

DRY WEIGHT	7000 lb
PROPELLANT CAPACITY	65,000 lb
GLOW	152,500 lb
PERIGEE KICK DELIVERY MODE	
PAYLOAD TO TRANSFER ORBIT	80,500 lb
USABLE PAYLOAD TO GEO	44,000 lb



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1590-037(T)

In the changeover of the 4-LO_2/LH_2 from a payload delivery vehicle to a manned service vehicle, the short vehicle nosecap is removed and replaced by a longer nosecap section. This allows a crew capsule to be contained inside the aeroshell and attached to the existing payload support structure at the forward bulkhead of the propulsion module of the 4-LO_2/LH_2. Also, the long nose provides additional changing L/D for a manned mission. Rotation open of the vehicle nose puts the vehicle in its servicing position. No deployment mechanisms are required to bring the fixed crew capsule to its servicing position. The support of the crew capsule by existing payload support structure, as well as the elimination of deployment mechanisms for the capsule, save weight and increase system reliability. A second advantage of the fixed capsule configuration is that it requires no breakup of propulsion subsystems in the changeover from delivery vehicle to manned vehicle. The propulsion system is entirely contained within the core module of the 4-LO_2/LH_2, as compared to the methods used on the 2-N_2O_4/MMH and 3-N_2O_4/MMH, allows the elimination of one structural interface. Based on both propulsion and structure considerations, the 4-LO_2/LH_2 type vehicle layout is superior to both the 2-N_2O_4/MMH and 3-N_2O_4/MMH layout when frequent changeovers from unmanned delivery vehicles to manned service vehicles are required.

The 4M-LO_2/LH_2, as shown in the figure, has a design driver mission of 14,000 lb delivery, 14,000 lb return, manned GEO service mission. The 55 ft, 14.5 ft diameter vehicle is shuttle transportable, but in order to perform its design driver mission, multiple shuttle launches or space basing is required.

4M – LO_2/LH_2 ORBITER TRANSPORT CONFIGURATION

1995 TECHNOLOGY AOTV

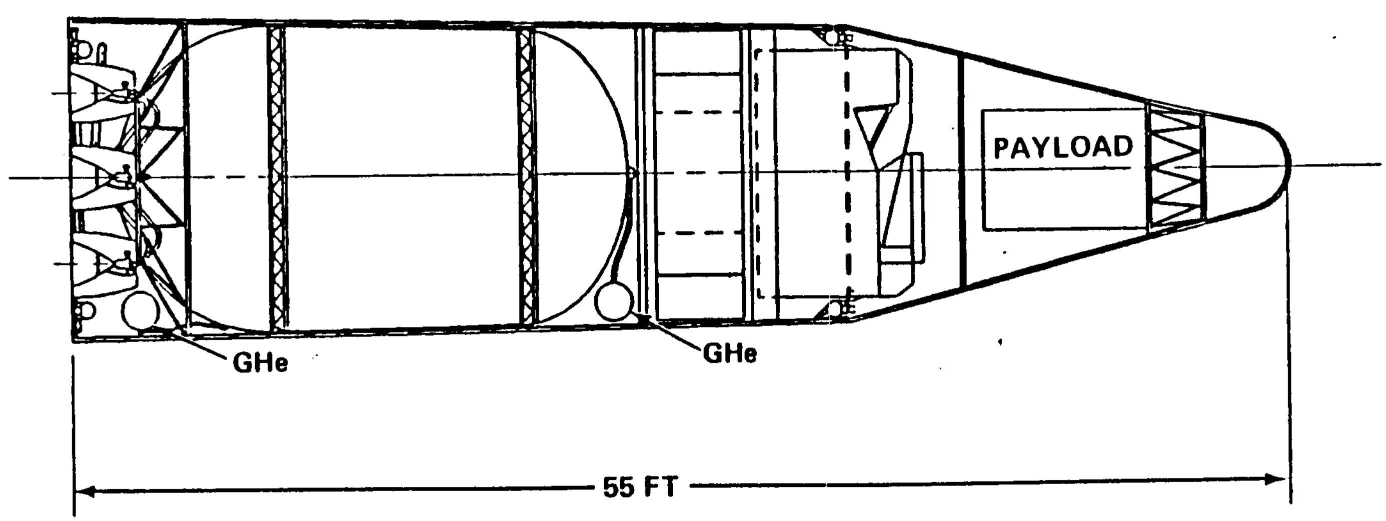
MANNED SERVICE VEHICLE

PAYLOAD CAPACITY = 14,000 lb DELIVERED AND RETURNED FROM GEO

SIX HINGED ENGINES: TOTAL THRUST = 18,000 lbf MR = 7/1

TOTAL PROPELLANT CAPACITY = 65,000 lb

I_{SP} = 480 lbf sec/lbm



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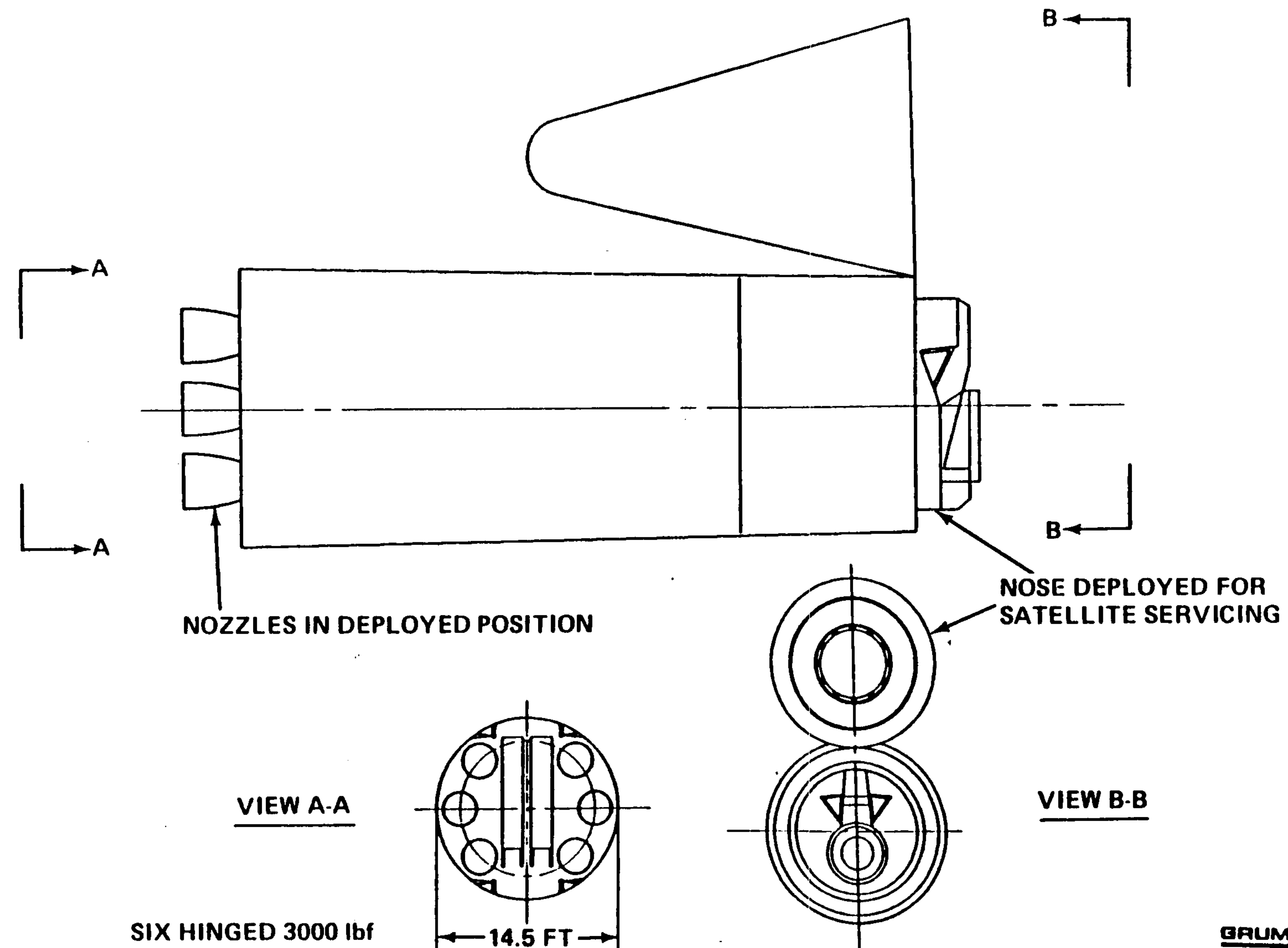
The 4M-LO_2/LH_2 with the nose deployed to a satellite servicing position is illustrated. The engines are shown in their deployed firing position.

4M – LO_2/LH_2 SPACE OPERATIONS CONFIGURATION



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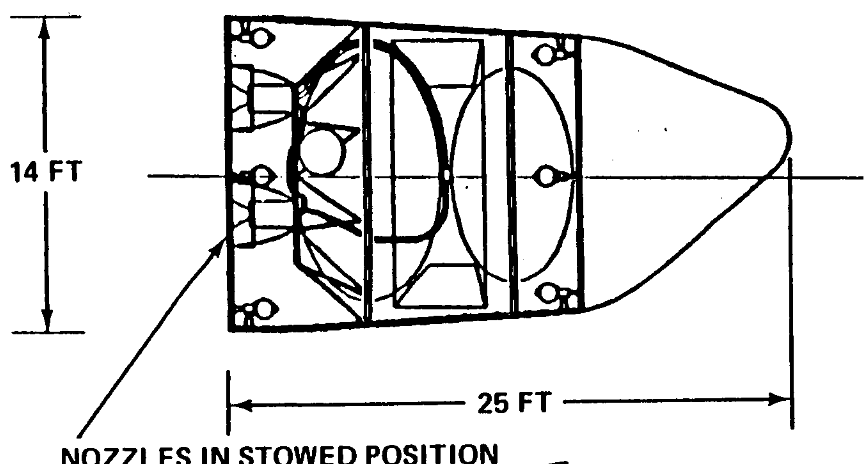
1590-017(T)

In order to evaluate the propellant combination of tetraflurohydrazine/hydrazine (N_2F_4/N_2H_4) as an alternate to LO_2/LH_2, vehicle concepts using N_2F_4/N_2H_4 as a propellant combination were generated. The vehicle shown on the facing page uses N_2F_4/N_2H_4 as propellant for its two piece deployable nozzle engines. The vehicle has internal tankage sized for an 8K delivery, 6K return, manned GEO mission. As shown in its payload delivery configuration, it is capable of delivering 28,400 lb of useful payload to GEO in a perigee kick mode. This weight is representative of 4 average commercial 1995-2000 payloads.

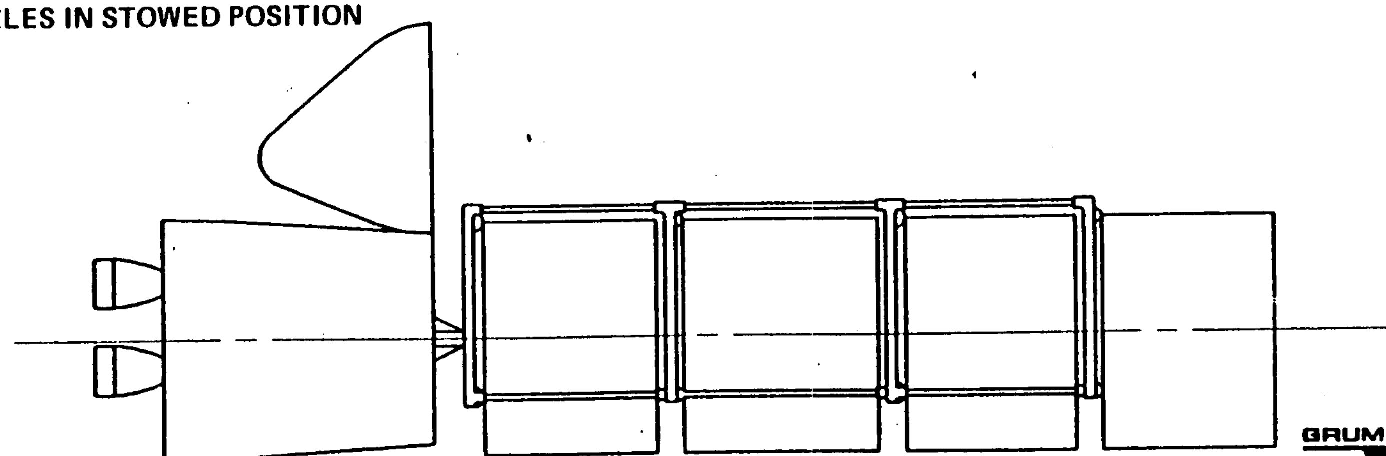
The compactness of the vehicle is the result of the use of deployable nozzles as well as the high specific impulse and high density impulse of this propellant combination. The propulsion systems (ACS/MPS) of this modular vehicle are all contained within one module of the vehicle. This allows easy transition from a payload vehicle to a manned service vehicle.

1 – N_2F_4/N_2H_4 MODULAR DELIVERY VEHICLE

- ADVANCED TECHNOLOGY VEHICLE
- VEHICLE TANKAGE SIZED FOR MANNED GEO SERVICE (8K UP & 6K BACK)
- THREE ENGINES TOTAL THRUST 18,000 lbf
- I_{SP} = 383 lbf sec/lbm



DRY WEIGHT	6800 lb
PROPELLANT CAPACITY	58,300 lb
GLOW	117,300 lb
PERIGEE KICK DELIVERY MODE	
• PAYLOAD TO TR.\NSFER ORBIT	52,200 lb
• USEABLE PAYLOAD TO GEO	28,400 lb



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In the transition from the unmanned 1-N_2F_4/N_2H_4 to the manned 1M-N_2F_4/N_2H_4, the delivery vehicle nose is removed and replaced with a longer deployable nose section. The crew capsule is coupled to the payload support structure of the vehicle. The propulsion core remains unchanged from the delivery vehicle configuration and requires no feed line (ACS/MPS) runs across the structural interface of the two modules. As compared to the storable vehicles presented earlier in this report (2-N_2O_4/MMH and 3-N_2O_4/MMH), one less structural interface, one less module, and no decoupling of propellant lines are required for the reconfiguration of the 1-N_2F_4/N_2H_4 into the 1M-N_2F_4/N_2H_4.

The change in gross lift off weight between the tetraflurohydrazine/hydrazine vehicles and the nitrogentetroxide/monomethylhydrazine vehicles is 30% of the lift off weight of the N_2O_4/MMH vehicle. This lower weight of the N_2F_4/N_2H_4 vehicles may be within the capabilities of enhanced orbiters. These are strong advantages of the tetraflurohydrazine/hydrazine vehicles over other storable propellant vehicles. Further study of N_2F_4/N_2H_4 seems warranted.

1M – N_2F_4/N_2H_4 MODULAR MANNED VEHICLE

ADVANCED TECHNOLOGY VEHICLE
THREE ENGINES TOTAL THRUST = 18,000 lbf
I_{SP} = 383 lbf sec/lbm

DRY WEIGHT	12,400 lb
PROPELLANT CAPACITY	58,300 lb
PAYLOAD LEFT AT GEO	1800 lb
GLOW	72,500 lb

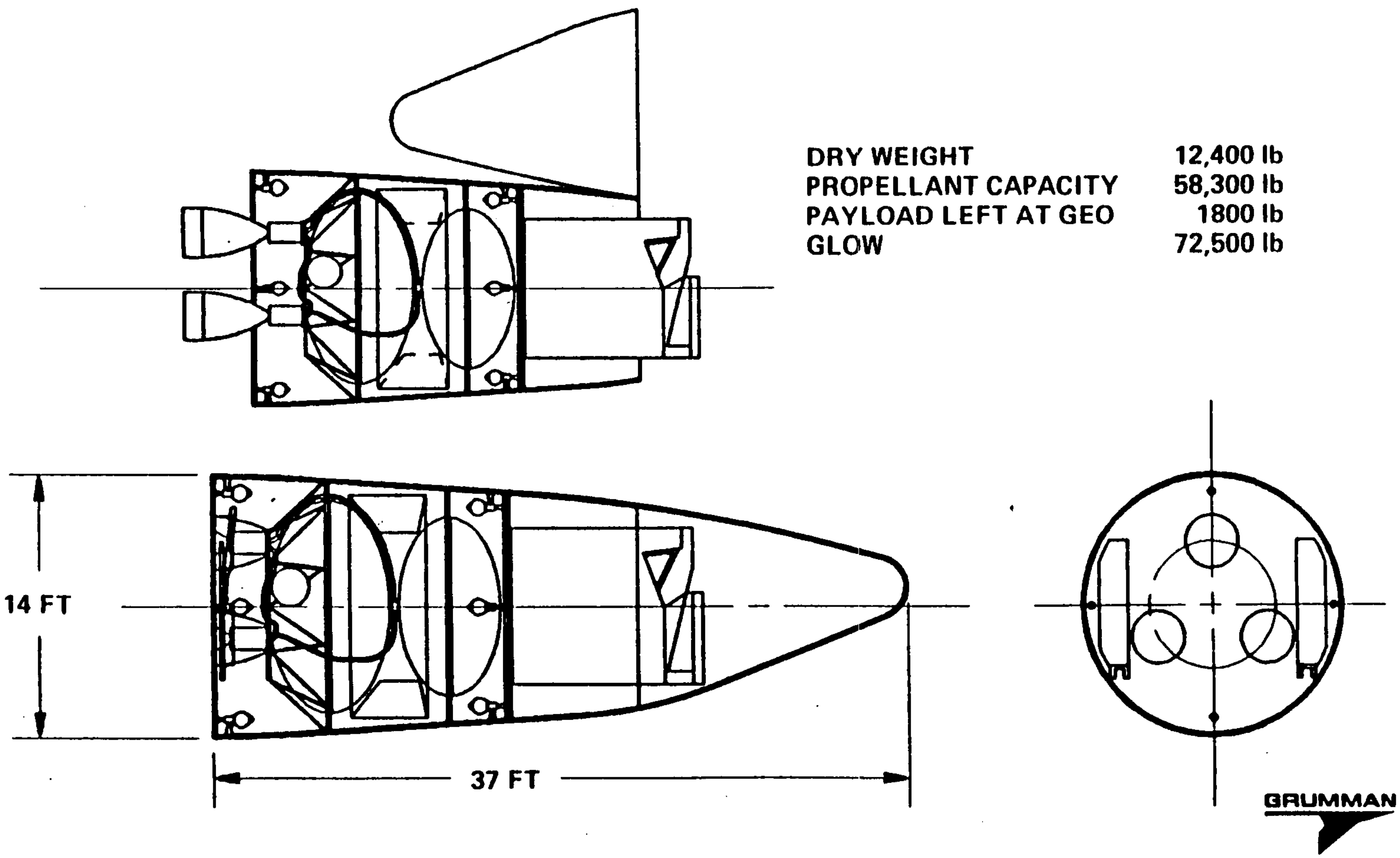

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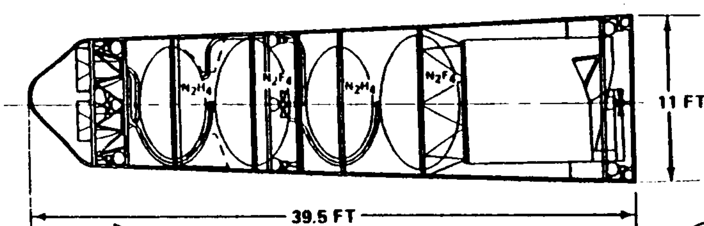
Figure A shows another unmanned cargo vehicle using N_2F_4/N_2H_4. Figure B shows the manned version of 3-N_2F_4/N_2H_4. The vehicle performs a 8k delivery 6K return manned GEO service with all internal tankage. It is totally reusable.

The configuration of this vehicle is different from the previously presented vehicles with fixed crew capsules. In the 3M-N_2F_4/N_2H_4 the tail deploys open for satellite service, the nose deploys open for engine firing and the auxiliary tankage required for the manned mission is inside the aeroshell. Two MPS propellant feed lines carry across the structural interface between the modules. At the vehicle tail, either flaps or ACS thrust may be used for aeroassist control. The configuration on Figure A and B is shown with the latter.

With a GLOW of 75,200 lbm for the manned vehicle, and 59,600 lbm for the delivery vehicle, the 3-N_2F_4/N_2H_4 family of vehicles is close to the capacity of a growth shuttle and may be used in ground based or space based operation scenarios.

3 – N_2F_4/N_2H_4 MODULAR DELIVERY VEHICLE

- ADVANCED TECHNOLOGY VEHICLE
- TOTALLY REUSABLE VEHICLE WITH AUXILIARY TANKS CONTAINED IN CREW CAPSULE
- THREE ENGINES TOTAL THRUST 18,000 lbf, I_{SP} = 383 lbf sec/lbm
- TWO FLUID CONNECTIONS ACROSS REMOVABLE INTERFACE



DRY WEIGHT	13,000 lb
PAYLOAD LEFT AT GEO	2100 lb
PROPELLANT CAPACITY	60,000 lb
MAIN TANKS	(29,500 lb)
AUX TANKS	(30,600 lb)
GLOW	75,200 lb

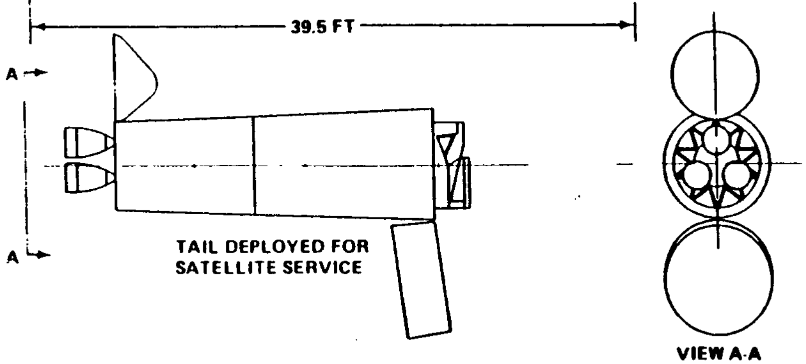
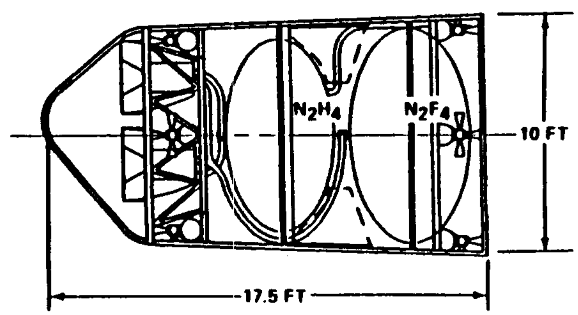


FIGURE B

- ADVANCED TECHNOLOGY VEHICLE
- VEHICLE TANKAGE SIZED FOR 13,200 lb OF USABLE PAYLOAD TO GEO IN PERIGEE KICK MODE
- THREE ENGINES TOTAL THRUST = 18,000 lbf
- I_{SP} = 383 lbf sec/lbm
- NO FLAPS
  - ACS USED FOR MANEUVERS IN ATMOSPHERE



DRY WEIGHT	5600 lb
PROPELLANT CAPACITY	24,500 lb
GLOW	59,600 lb
PERIGEE KICK DELIVERY MODE	
• PAYLOAD TO TRANSFER ORBIT	24,500 lb
• USEABLE PAYLOAD AT GEO	13,200 lb

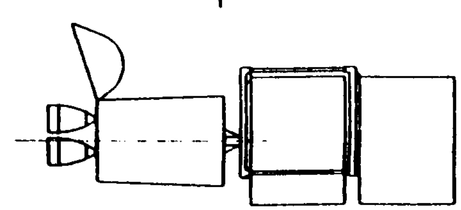


FIGURE A

The conclusions we have reached from studying our modular vehicle concepts are summarized.

SOME CONFIGURATION CONCLUSIONS

- LIMITING INTERNAL TANKAGE CAPACITY TO THE NEEDS OF MOST SATELLITE DELIVERY MISSIONS PROVIDES THE MOST FLEXIBLE AOTV
  - PRESERVES GROUND LAUNCH CAPABILITY
  - PAYLOAD MANIFESTING CLOSER TO CURRENT PRACTICE
- OPENING AOTV FOR INSERTION OF MANNED MODULE ENHANCES SPACE-BASED MAINTENANCE
  - PERMITS ACCESS NEAR ℄ OF VEHICLE
- SINGLE STS FLIGHT GROUND-BASED MANNED MISSION TO GEO POSSIBLE WITH ENHANCED ORBITER (77K CARGO) AND N_2F_4/N_2H_4 POWERED AOTV
- PREFERRED MANNED CAPSULE INSTALLATION KEEPS CAPSULE FIXED TO AOTV
  - MANNED ACCESS THROUGH ROTATING NOSE
  - NO DISRUPTION OF FLUID LINES DURING VEHICLE CHANGEOVER
  - REDUCED NUMBER OF STRUCTURAL LATCHING INTERFACES

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3.2.3 Mid L/D AOTV/Payload Manifesting

An overview of the manifesting study methodology and results is outlined in the figure.

MANIFESTING STUDY OVERVIEW

- METHODOLOGY
  - — ANALYSIS, QUANTIFICATION & AUGMENTATION OF MID 1984 MISSION MODEL OF SPACE STATION WORKING GROUP
    - SELECTED 6 YEAR PERIOD
  - — USED PROPRIETARY COMPUTER PROGRAM, "GOOD FIT", TO FIT ALL SPACE BOUND ($0 \leq i \leq 30°$) NASA PAYLOADS INTO THE CARGO BAY OF THE ORBITER AND COUNT THE NUMBER OF SHUTTLE FLIGHTS TO SPACE STATION
- SUMMARY OF RESULTS & RECOMMENDATIONS
  - — FOR SB LOX-H AOTVs, LARGE AOTV (14K UP & BACK) AND "NOMINAL" PAYLOAD (13.2K) PK AOTV REQUIRE SAME NUMBER OF STS FLIGHTS (85). ALSO, LARGE AOTV REQUIRES 4 FEWER OTV FLIGHTS
    - RECOMMENDATION: DESIGN FULL SIZE SPACE BASED AOTVs
  - — COMPARED SPACE BASED OPERATION OF 4 PAIRS OF OTVs (WITH 4 PROPELLANTS), ONE OPTIMIZED FOR WEIGHT (SB), THE OTHER OPTIMIZED FOR LENGTH (GB)
    - SB SAVE AN AVERAGE OF 1 STS FLIGHT (MAX OF 2) OVER 6 YEARS
    - RECOMMENDATION: ALL AOTVs SHOULD BE STRONG ENOUGH TO OPERATE IN GB MODE
  - — COMPARED GB OTV OPERATION WITH 4 DIFFERENT PROPELLANTS: LOX-HYDROGEN & 3 "STORABLES" (N_2O_4/MMH, LOX/MMH, N_2F_4/N_2H_4)
    - COMPUTER ANALYSIS CONTAINED ONLY 176 PAYLOADS
      - — 3 ESCAPE PAYLOADS NOT INCLUDED
    - ALL STORABLES REQUIRED ≤ 90 STS FLIGHTS, 3 FEWER THAN 93 REQ'D BY LOX/H_2
    - CURRENT RECOMMENDATION: GB CARGO AOTV SHOULD USE N_2O_4/MMH
  - — COMPARED SB AOTV OPERATION WITH 4 DIFFERENT PROPELLANTS
    - COMPUTER ANALYSIS CONTAINED 176 PAYLOADS
      - — 3 ESCAPE PAYLOADS NOT INCLUDED
    - LOX-HYDROGEN HAD THE LOWEST NUMBER OF STS FLIGHTS (86) THEN N_2F_4/N_2H_4 (92), THEN LOX/MMH (94), FINALLY N_2O_4/MMH (98)
    - ROM CRUDE ECONOMIC ANALYSIS INDICATES ALL PROPELLANTS COST THE SAME FOR THESE 6 YEARS
      - — FURTHER WORK NEEDED HERE
    - CURRENT RECOMMENDATION: SB CARGO AOTV SHOULD USE N_2O_4/MMH
      - — QUALIFICATION: MANNED MISSIONS, WHICH BENEFIT FROM SMALL, HIGH I_{SP} LOX/H_2 ENGINES, WERE NOT CONSIDERED IN THE MANIFESTING STUDY

- SUMMARY RESULTS & RECOMMENDATIONS FOR SS TECHNOLOGY:
  - — FOR GB AOTV, COMPARED GB & SB PAYLOAD MANIFESTING
    - SB PAYLOAD MANIFESTING SAVES 1 TO 3 STS FLIGHTS OVER 6 YEARS
    - RECOMMENDATION: WHEN NECESSARY, SPACE STATION SHOULD PERFORM PAYLOAD MANIFESTING OF GB AOTV PAYLOADS
  - — FOR GROUND MAINTAINED AOTV (UP & BACK TO EARTH FOR EACH OTV FLT), COMPARED GB VS SPACE BASED MANIFESTING OF PAYLOADS & PROPELLANTS FOR DIFFERENT PROPELLANTS
    - SPACE BASING SAVES 6 OR 7 STS FLTS OVER 6 YEARS
    - RECOMMENDATIONS: SINCE BENFIT OF SS HANDLING OF PROPELLANTS IS FUNCTION OF DEVELOPMENT COSTS

      LO_2/LH_2 — NOT RECOMMENDED FOR SS HANDLING

      N_2O_4/MMH — RECOMMENDED FOR SS HANDLING

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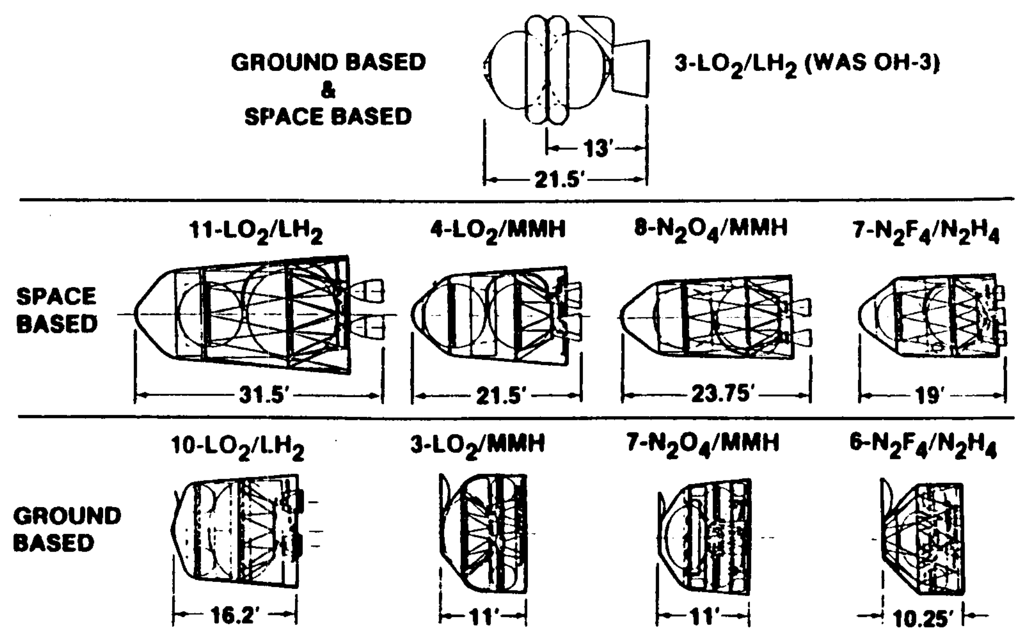
The vehicle configuration shown in the following figures are concepts generated for use in the OTV propellant/basing mode/delivery mode tradeoff study utilizing the computer manifesting program, "Good Fit". All vehicle concepts are man-rated cargo vehicles sized for the delivery of 13,200 lb of usable payload to GEO. Four different propellant combinations are represented in both ground-based and space-based mode vehicles. The concepts were produced using available vendor data on engine size/performance. These concepts are not optimized designs, but are very representative of what can be expected of a similar class vehicle (same design mission, basing mode, propellant combination). The vehicles provide a data base for the mass properties, geometry, and main propulsion system performance that are used in the manifesting program.

All the vehicles evaluated in the manifesting study are shuttle transportble. Space-based vehicles are weight optimized with light structural shell weights, an advanced technology nozzle protection regime which shortens and lightens the aeroshell, and lightweight propellant tankage. These space-based vehicles are not compatible with ground based operations because the lightweight structure of the vehicle is not capable of surviving a shuttle launch when the OTV propellant tanks are loaded. Secondly, an upgraded purge/insulation system would be required for ground based operation on the space base vehicles.

Ground based vehicles in the study are optimized for length. The use of heavy torrodial tanks is justified because it allows a better packaging of propellant and therefore a much shorter vehicle. This weight/length tradeoff is justified for ground basing because it allows the delivery of many NASA GEO payloads (based on mid-summary 1984 NASA mission model 1995-2000) that would be excluded from single shuttle launch delivery if the ground-based OTV were as long as comparable space based OTV. The ground based OTV's have structural and thermal control systems compatible with fully tanked OTV/shuttle launches.

VEHICLE CONFIGURATIONS

- VEHICLES STUDIED WITH COMPUTERIZED MANIFESTING STUDY
  - ALL VEHICLES ARE MAN-RATED CARGO VEHICLES SIZED FOR PERIGEE KICK DELIVERY OF 13,200 lb OF USEFUL CARGO AT GEO
  - 9 CONFIGURATIONS
  - 4 DIFFERENT PROPELLANT SYSTEMS
  - 5 LENGTH OPTIMIZED GROUND BASED VEHICLES
  - 4 WEIGHT OPTIMIZED SPACE BASED VEHICLES
- DETAILS OF NEW VEHICLE DESIGNS
  - MAJOR CHARACTERISTICS
    - SIZE
    - WEIGHT
    - PROPULSION
  - WEIGHT STATEMENTS



- SPACE BASED VEHICLES ARE OPTIMIZED FOR MINIMUM WEIGHT
- GROUND BASED VEHICLES ARE OPTIMIZED FOR MINIMUM LENGTH

Two new LO_2/LH_2 OTV's are shown in the Figure. The comparison of vehicle lengths and weight statements expose the differences between space-based and ground-based vehicles. Space based vehicles are weight optimized vehicles incorporating lightweight shell structure while ground based vehicles use heavier structure which is compatible with shuttle delivery to LEO with full OTV propellant tanks. This difference in structural weight is revealed in the weight statement comparison shown in Figure B21. The shorter vehicle weighs 575 lb (+9%) mroe than the longer vehicle.

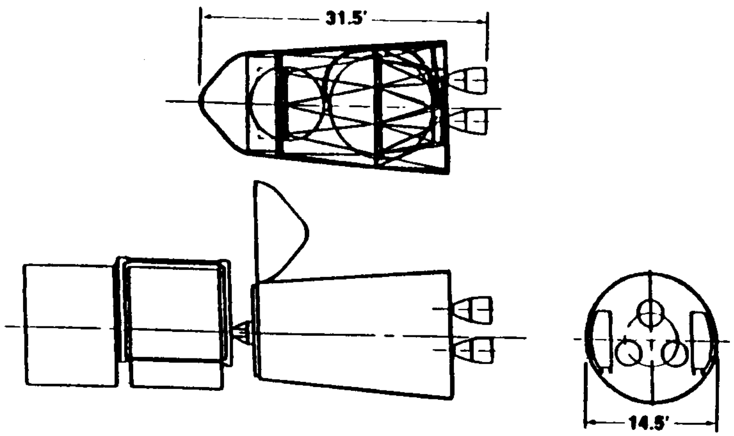
Additional structural weight is required in the propellant tankage for the transition between the space-based OTV and the ground-based OTV. This weight increase is caused by the required strengthening of the tanks for ground based operations as well as an increase due to going to more structurally inefficient propellant tankage (spherical to torrodial on the oxidizer tank) which is required to shorten the vehicle to make it compatible with ground based operation.

The third and final area which reflects a weight difference is the insulation/purge system, where the more stringent ground based vehicle requirements produce the differences shown in the weight statement (Page).

The composite of all these factors cause an increase in weight of approximately ten percent from the space based vehicle to the ground based vehicle.

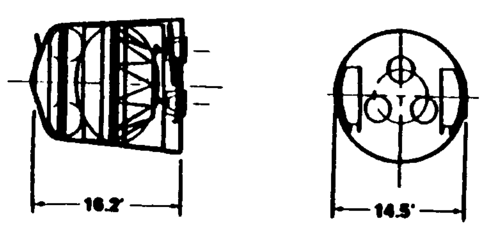
NEW SPACE BASED OTV: 11-LO_2/LH_2

DRY WEIGHT	=	5725 lbm
PROPELLANT CAPACITY	=	22550 lbm at MR = 6/1
TOTAL THRUST	=	9000 lb_f
I_{SP}	=	480 $lb_f \cdot sec/lbm$
X_{CM}	=	.48L



NEW GROUND BASED OTV: 10-LO_2/LH_2

DRY WEIGHT	=	6300 lbm
PROPELLANT CAPACITY	=	22550 lbm at MR = 6/1
TOTAL THRUST	=	9000 lb_f
I_{SP}	=	480 $lb_f \cdot sec/lbm$
X_{CM}	=	.58L



WEIGHT STATEMENTS FOR LOX-HYDROGEN AOTVs

	SPACE BASED 11-LO_2/LH_2		GROUND BASED 10-LO_2/LH_2	
STRUCTURE		1350		1845
SHELL & SUPPORT STRUCTURE	300		395	
FUEL TANK	220		300	
OXIDIZER TANK	280		600	
FLAPS	350		350	
BERTHING, DOCKING & PAYLOAD SUPPORT	200		200	
THERMAL PROTECTION SYSTEM		825		850
SHELL TPS	690		430	
TANK TPS	135		420	
PROPULSION SYSTEM		900		900
ENGINES	330		330	
PLUMBING & GIMBAL DRIVES	570		570	
ACS	450		450	
EPS	980		980	
AVIONICS	700		700	
SUBTOTAL	5205		5725	
+10%				
DRY WEIGHT	5725		6300	

Δ WEIGHT = 575 LB = 9%

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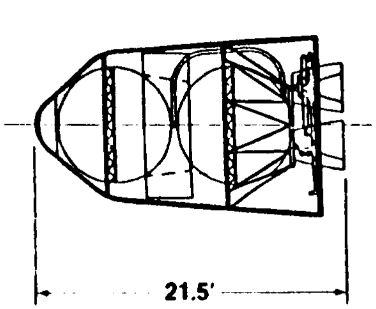
An alternative propellant combination to LO_2/LH_2 that was explored in the computer manifesting tradeoff study was Liquid Oxygen/Monomethyl Hydrazine (LO_2/MMH). Two vehicles using this propellant combination, one space based, the other ground based, are shown in the figure. These vehicles which were evaluated by the program "Good Fit", are designed to the same reference mission as all the other internally tanked vehicles shown on Page . The reference mission is delivery of 13,200 lbm usable payload at GEO via perigee kick from the AOTV. The advanced engine performance/size data (for the LO_2/MMH propellants) used in this vehicle sizing was supplied by Rocketdyne.

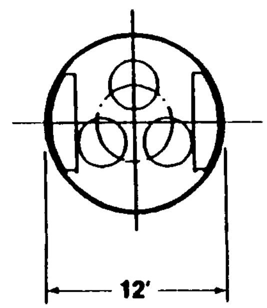
The propellant combination LO_2/MMH represents a compromise between space storable N_2O_4/MMH and high performance cryogenic oxygen and hydrogen. The LO_2/MMH provide substantially cheaper space storage than LO_2/LH_2 and higher I_{SP} than N_2O_4/MMH. The characteristics of both LO_2 and MMH are known and there is an experience base for each. Disadvantages to LO_2/MMH are the relatively poor density impulse and the fact that no advanced engine technology development program is in place for LO_2/MMH.

The weight statement comparison between the space based and ground based LO_2/MMH vehicles is also shown in the Figure.

NEW SPACE BASED OTV: 4-LO_2/MMH

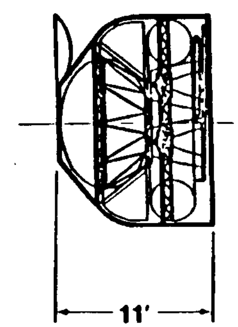
DRY WEIGHT	=	5375 lbm
PROPELLANT CAPACITY	=	32100 lbm at MR = 1.41/1
TOTAL THRUST	=	18000 lb_f
I_{SP}	=	373 $lb_f \cdot$ sec/lbm
X_{CM}	=	.55L

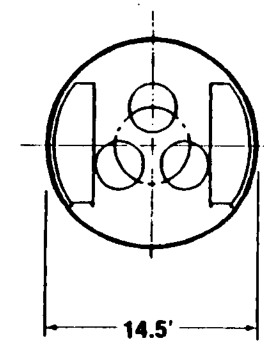




NEW GROUND BASED OTV: 3-LO_2/MMH

DRY WEIGHT	=	5950 lbm
PROPELLANT CAPACITY	=	32100 lbm at MR = 1.4/1
TOTAL THRUST	=	18000 lb_f
I_{SP}	=	373 lb_f sec/lbm
X_{CM}	=	.55L





WEIGHT STATEMENTS FOR LOX-MMH AOTVs

	SPACE BASED 4-LO_2/MMH		GROUND BASED 3-LO_2/MMH	
STRUCTURE		1215		1840
SHELL & SUPPORT STRUCTURE	255		290	
FUEL TANK	195		600	
OXIDIZER TANK	215		410	
FLAPS	350		350	
BERTHING, DOCKING & PAYLOAD SUPPORT	200		200	
THERMAL PROTECTION SYSTEM		515		380
SHELL TPS	465		320	
TANK TPS	50		60	
PROPULSION SYSTEM		1105		1130
ENGINES	425		425	
PLUMBING & GIMBAL DRIVES	680		705	
ACS	450		450	
EPS	900		900	
AVIONICS	700		700	
SUBTOTAL	4885		5410	
+10%				
DRY WEIGHT	5375		5950	

Δ WEIGHT = 575 LB = 10%

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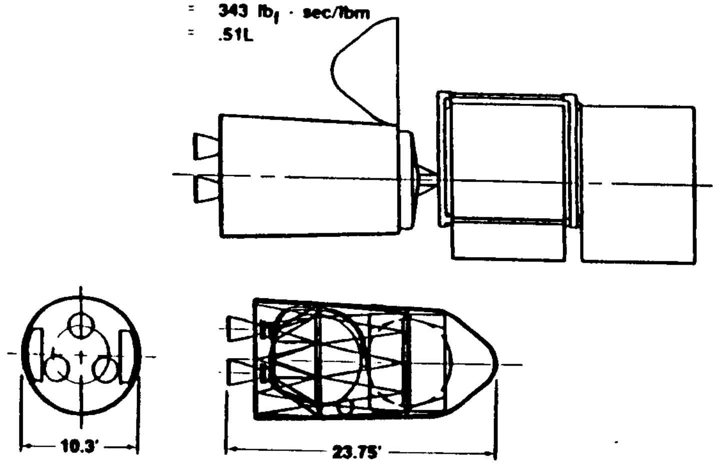
N_2O_4/MMH offers many advantages over the other propellants we have studied. This propellant is space storable, familiar, common to the OMV, and has an engine development program well underway.

Two vehicles using this propellant combination, one space-based, the other ground based, are shown in the figure and were evaluated in the computer manifesting study. Near term advanced Aerojet Transtar III or Rocketdyne XLR-132 class engines, were used for primary propulsion. They both represent low start up cost advanced engines that, with funding for adaptation to allow multiple reuse, would be available for a near term OTV.

The weight statement for the two vehicles, shows the difference associated with a purely space based vehicle and a ground based vehicle. The 10% difference is consistent with the 9-10% deviation associated with the space to ground transition of vehicles using the other propellant combinations considered in this manifesting study.

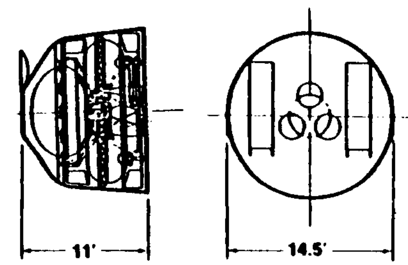
NEW SPACE BASED OTV: 8-N_2O_4/MMH

DRY WEIGHT	= 5150 lbm
PROPELLANT CAPACITY	= 35300 lbm at MR = 2.1/1
TOTAL THRUST	= 11250 lb_f
I_{SP}	= 343 $lb_f \cdot sec/lbm$
X_{CM}	= .51L



NEW GROUND BASED OTV: 7-N_2O_4/MMH

DRY WEIGHT	= 5700 lbm
PROPELLANT CAPACITY	= 35300 lbm at MR = 2.1/1
TOTAL THRUST	= 11250 lb_f
I_{SP}	= 343 $lb_f \cdot sec/lbm$
X_{CM}	= .54L



WEIGHT STATEMENTS FOR NITROGENTETRAOXIDE/MMH AOTVs

	SPACE BASED 8-N_2O_4/MMH		GROUND BASED 7-N_2O_4/MMH	
STRUCTURE		1205		1770
SHELL & SUPPORT STRUCTURE	245		310	
FUEL TANK	180		310	
OXIDIZER TANK	250		600	
FLAPS	350		350	
BERTHING, DOCKING & PAYLOAD SUPPORT	200		200	
THERMAL PROTECTION SYSTEM		460		375
SHELL TPS	440		340	
TANK TPS	20		35	
PROPULSION SYSTEM		965		985
ENGINES	315		315	
PLUMBING & GIMBAL DRIVES	650		670	
ACS	450		450	
EPS	900		900	
AVIONICS	700		700	
SUBTOTAL	4680		5180	
+10%				
DRY WEIGHT	5150		5700	

Δ WEIGHT = 550 LB = 10%

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GRUMMAN

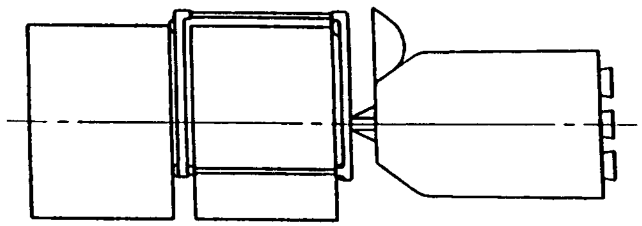
Tetraflurohydrazine/Hydrazine appears as an attractive propellant because of its high specific impulse and density impulse. Vehicles using this "semi-cryogenic" propellant combination were generated for evaluation in our manifesting tradeoff study.

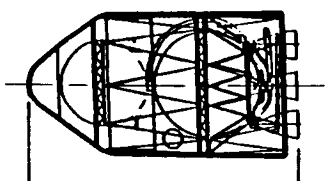
Both the space-based and ground-based vehicle using tetraflurohydrazine/hydrazine are the smallest geometrical packages of any of the vehicles considered in this tradeoff. The particularly short ground based 6-N_2F_4/N_2H_4 achieved its diminutive size by taking advantage of nesting both the fuel tank and engines inside the oxidizer torus. The weight of the two N_2F_4/N_2H_4 vehicles is slightly higher than would be expected for such small vehicles. This higher weight is caused by a material incompatibility problem associated with the oxidizer, tetraflurohydrazine. Stainless steel is required for the pressure vessels, lines and fittings which contact the oxidizer.

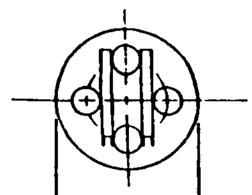
Other disadvantages of tetraflurohydrazine with respect to an advanced AOTV program are limited experience in handling tetraflurohydrazine and that no engine development program is underway for this propellant combination.

NEW SPACE BASED OTV: 7-N_2F_4/N_2H_4

DRY WEIGHT	=	5280 lbm
PROPELLANT CAPACITY	=	29220 at MR = 3.07/1
TOTAL THRUST	=	12000 lb_f
I_{SP}	=	388 $lb_f \cdot$ sec/lbm
X_{CM}	=	.52

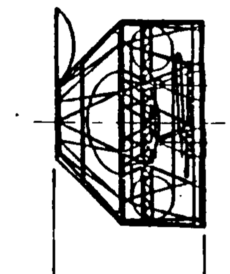


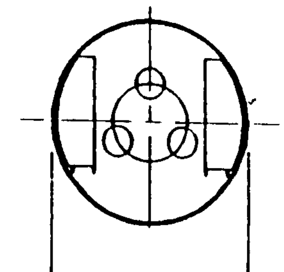




NEW GROUND BASED OTV: 6-N_2F_4/N_2H_4

DRY WEIGHT	=	5860 lbm
PROPELLANT CAPACITY	=	29400 lbm at MR = 3.07/1
TOTAL THRUST	=	9000 lb_f
I_{SP}	=	388 $lb_f \cdot$ sec/lbm
X_{CM}	=	.51





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0283-023PP

WEIGHT STATEMENTS FOR TETRAFLUOROHYDRAZINE-HYDRAZINE AOTVs

	SPACE BASED 7-N_2F_4/N_2H_4		GROUND BASED 6-N_2F_4/N_2H_4	
STRUCTURE		1295		1705
SHELL & SUPPORT STRUCTURE	195		265	
FUEL TANK	170		240	
OXIDIZER TANK	380		650	
FLAPS	350		350	
BERTHING, DOCKING & PAYLOAD SUPPORT	200		200	
THERMAL PROTECTION SYSTEM		400		490
SHELL TPS	350		285	
TANK TPS	50		205	
PROPULSION SYSTEM		1055		1080
ENGINES	380		380	
PLUMBING & GIMBAL DRIVES	675		700	
ACS	450		450	
EPS	900		900	
AVIONICS	700		700	
SUBTOTAL	4800		5325	
+10%				
DRY WEIGHT	5280		5860	

Δ WEIGHT = 580 LB : 10%

GRUMMAN

The mission model used for Grumman"s "Good Fit" was derived from the summer 1984 mission model issued by NASA's Space Station working group and covers space station, space station orbit, GEO, and earth escape trajectory payloads. Additionally the Grumman OTV mission model includes delivery to space station of the space station logistics modules, OMV propellant, GEO based OMV propellant and vehicles, and OTV's used in their various operational modes. The model currently covers a 6 year period, 1995-2000, and consists of 179 payloads.

MANIFESTING METHODOLOGY: MISSION MODEL

- SUMMER, 1984 NASA MISSION MODEL
  - SATELLITES/EXPERIMENTS FOR SPACE STATION, NEAR SPACE STATION ORBITS & GEO
  - EXPLORATION SATELLITES FOR EARTH ESCAPE TRAJECTORIES
- ADDITIONS TO NASA MODEL
  - LOGISTICS MODULES FOR SPACE STATION
  - OMV PROPELLANT FOR SPACE STATION
  - OMV PROPELLANT & VEHICLES FOR GEO PLATFORM
  - OTV PROPELLANT & VEHICLES FOR GB & GROUND MAINTAINED MODES
  - OTV PROPELLANT AND SUPPLEMENTARY TANKS FOR SB MODES
- ADJUSTMENTS TO MISSION MODEL
  - OCCASIONAL VERY LONG OR VERY HEAVY PAYLOAD DIVIDED INTO TWO EQUAL PAYLOADS
  - QUANTIFY (WEIGHT & LENGTH) VAGUE PAYLOADS
- SELECTED MODEL
  - 1995 TO 2000 (6 YEARS)
  - TOTAL OF 179 PAYLOADS (~30 PAYLOADS/YEAR)

GRUMMAN

0283-030PP

Good Fit is a computer program for manifesting both the orbiter and AOTV's. The program is capable of stacking the STS cargo bay based on the length, weight, and CG constraints of the shuttle. The program also manifests and calculates propellant requirements for OTV delivery of payloads to high energy orbits.

The manifesting is accomplished by first defining the characteristics (length, weight, CM location) of the payloads along with the time period in which they need to fly and their ultimate destination. This definition takes place inside the mission model data set. User defined priority factor are input interactively. Priority factors are maximum number of payloads per OTV flight, minimum acceptable propellant utilization for the OTV, maximum number of payloads per shuttle flight and minimum acceptable length/weight utilization per shuttle flight.

There is a choice of one of 4 basing modes available in the program. An explanation of each mode is given in the figure. Another choice exists between the two insertion modes, perigee kick and single stage. The entire mission model is run for one user selected OTV, utilizing the selected basing and insertion modes. Payloads requiring single stage insertion, when flown inside the perigee kick insertion mode, are still delivered in their required single stage insertion mode. The output of the program is a chronological listing of orbiter flights and OTV flights, with their manifested payloads. Propellant and OTV drop tanks, which may be empty or full, are manifested along with the payloads. Finally, summary for total shuttle flights, shuttle utilization, OTV flights and OTV utilization are presented as output after program execution.

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MANIFESTING METHODOLOGY: "GOOD FIT"

- "GOOD FIT" IS A PROPRIETARY COMPUTER PROGRAM WHICH WAS DESIGNED TO PERFORM MANIFESTING STUDIES FOR STS & OTV
- CHARACTERISTICS OF "GOOD FIT"
  - PAYLOADS ARE GROUPED BY ORIGIN, DESTINATION & TIME INTERVAL (E.G., 1 YEAR)
    - A PAYLOAD MUST FLY WITHIN ITS TIME INTERVAL (TI) OR ON THE FIRST FLIGHT OF THE NEXT TI
  - PAYLOADS MEET ALL ORBITER CARGO BAY CONSTRAINTS
    - LENGTH, WEIGHT & CENTER OF MASS LOCATION
  - MAXIMUM NUMBER OF PAYLOADS PER AOTV FLIGHT ARE FIXED PER RUN
    - ONE RUN COVERS MISSION MODEL
  - SELECT 1 OF 4 OTV BASING MODES
    - 1 = GROUND BASED OTV & ALL SUPPORTING OPS
    - 2 = GROUND BASED OTV WITH PAYLOAD STORAGE & MANIFESTING AT SPACE STATION
    - 3 = GROUND MAINTAINED OTV (RETURN TO EARTH AFTER EACH FLIGHT) WITH SPACE BASED SERVICES: PAYLOAD MANIFESTING & PROPELLANT STORAGE/DISPENSING
    - 4 = SPACE BASED OTV (OTV REMAINS IN SPACE)
- CHARACTERISTICS (CONT)
  - SELECT 1 OF 2 OPERATING MODES
    - 1 = PERIGEE KICK: ALL SATELLITES (EXCEPT OMV) WITH ON BOARD PROPULSION PERFORM APOGEE INSERTION BURN WITHOUT OTV. LENGTH & WEIGHT OF SATELLITE HAVE BEEN ENLARGED FOR ORBITER PACKAGING TO ACCEPT THIS EXTRA PROPELLANT
      - ALL OTHER OTV PAYLOADS ARE DELIVERED TO THEIR FINAL DESTINATION BY OTV
    - 2 = SINGLE STAGE INSERTION: ALL PAYLOADS ARE DELIVERED TO THEIR FINAL DESTINATIONS BY OTV
    - EXCEPTIONS TO ABOVE ARE EARTH ESCAPE PAYLOADS. REUSEABLE OTV DELIVERS PAYLOAD TO I = 28½° ESCAPE TRAJECTORY & THEN OTV RETURNS TO LEO
      - SUBROUTINE NOT OPERABLE AT THIS TIME
  - SELECT ANY OTV IN DATA FILE
  - PROGRAM SELECTS A PAYLOAD BASED ON PRIORITY FACTORS (INPUT VARIABLES) OF WEIGHT OR LENGTH
  - OUTPUT LISTS MANIFESTING FOR EACH ORBITER & OTV FLIGHT
    - PAYLOADS, PROPELLANT, EMPTY (OR FULL) DROP TANKS

GRUMMAN

The Grumman Computer Manifesting Program, "Good Fit", performs a shuttle manifest based on user input priority factors. Shuttle packaging which meets program user specified minimum utilization factors are deemed acceptable shuttle flights by the program. No attempt has been made to implement a logic for comparison of the computer selected manifest to all other possible manifests in order to arrive at an optimum manifest. Although the program produces a good manifest, it is probable that a manifest done by hand will achieve superior utilization of the shuttle and OTV. Further study into the optimization of shuttle manifesting is recommended.

COMMENTS ON "GOOD FIT"

- "GOOD FIT" DOESN'T PRODUCE BEST FIT, DOES PRODUCE A GOOD FIT
  - NOT AN OPTIMIZATION PROGRAM
  - DOES NOT COMPARE THE MANIFESTING IT HAS SELECTED WITH OTHER POSSIBLE MANIFESTS
- OPERATOR CONTROL OVER MANIFESTING BY ADJUSTING
  - PRIORITY FACTORS (FOR PAYLOAD WEIGHT & LENGTH)
  - PAYLOAD TIME INTERVALS & THE PAYLOADS THEY CONTAIN
  - MISSION MODEL
- WITHIN ABOVE CONSTRAINTS, "GOOD FIT" SELECTS PAYLOAD GROUPS FOR EACH FLIGHT
- HUMAN JUDGEMENT DURING MANIFESTING MAY IMPROVE RESULTS
  - TBD. THIS DESERVES ADDITIONAL STUDY

GRUMMAN

0283-032PP

The conditions applied to the computer manifest performed by "Good Fit" are reported here. The majority of the runs were done with an AOTV utilizing perigee kick delivery. Payloads unable or unwilling to perform their own apogee burn (e.g., a "payload" for a GEO platform, which does not have its own propulsion system) are delivred in single stage insertion. The four basing modes available in the program: ground based OTV and all supporting OPS, ground based OTV with space based payload storage and manifesting at the space station, ground based OTV with space based services and propellant storage, exclusive space basing, were examined in the computer runs. The results presented in this report are based on the assumption that a maximum of four payloads can be manifested on a single OTV flight.

"GOOD FIT" INITIAL CONDITIONS FOR THIS STUDY

- EARTH TO LEO TRANSPORT VIA STS ORBITER
  - MAX CARGO WEIGHT = 65,000 LB
  - MAX CARGO LENGTH = 60 FT
  - STANDARD CARGO CG LOCATION CONSTRAINT
  - ALL PAYLOADS GO TO SPACE STATION

- SPACE STATION TO GEO TRANSPORT VIA AOTV
  - OVER 12 DIFFERENT AOTV DESIGNS EXAMINED
    - LO_2/LH_2 WITH I_{SP} = 480 SEC & 475 SEC
    - LO_2/MMH WITH I_{SP} = 373 SEC
    - N_2O_4/MMH WITH I_{SP} = 343 SEC
    - N_2F_4/N_2H_4 WITH I_{SP} = 388 SEC
  - MOST RUNS USED PERIGEE KICK OPERATING MODE
  - ALL 4 BASING MODES EXAMINED
  - MAXIMUM OF 4 PAYLOADS PER OTV FLIGHT

GRUMMAN

0283-033(1/2)PP

The six year Grumman OTV mission model contained 176 payloads which ran in the computer program "Good Fit". They were divided into 3 time interval breakdowns: one year, half year, and quarter year intervals. The program attempts to fly all payloads from the same time interval in adjacent flights. For the Grumman study, the mission model was divided into three month intervals. This was chosen as the most reasonable because a payload has to be available at the start of a time interval, and may wait around to the end of the interval before it can be efficiently manifested. To ask many expensive payloads to sit on the shelf for 6 months or a year until the Shuttle was ready for them seem unrealistic.

"GOOD FIT" INITIAL CONDITIONS FOR THIS STUDY (CONT)

- MISSION MODEL
  - 6 YEARS (1995 TO 2000)
  - 179 PAYLOADS (3 ESCAPE TRAJECTORY PAYLOADS DIDN'T RUN)
  - EXAMINED 3 TIME INTERVALS
    - 12 MONTHS → TOO LONG
    - 6 MONTHS
    - 3 MONTHS → MOST REASONABLE FOR CUSTOMER & CHOSEN FOR THIS STUDY

GRUMMAN

0283-033(2/2)PP

A manifesting study tradeoff was performed to determine an optimum size for a Space Based AOTV for delivering the Grumman augmented mid summer 1984 NASA mission model. Two space based vehicles were selected for this comparison, the first vehicle, 4-LO_2/LH_2, is a vehicle with internal tankage sized for a 14K up and back manned GEO Service mission. The second vehicle is the 11-LO_2/LH_2 (Page), whose internal tankage is sized for a 13,200 lb usable payload at GEO delivery mission in perigee kick mode.

Using the Good Fit program, the two vehicles were run through the 1995-2000 Grumman OTV mission model with identical operating modes and utilization factors. Both vehicles were able to complete the mission model in 85 STS flights, although the smaller vehicle required four more AOTV flights to deliver all GEO payloads. The expected performance advantage of the smaller lighter vehicle was insignificant when considering the packaging of STS flights. This is explained by the fact that the extra propellant required by the larger vehicle on the majority of payload delivery missions is of a small enough quantity that it may be packaged in the cargo bay along with other LEO and GEO payloads without the addition of any extra STS flights. Also, the large vehicle has a higher thrust/weight, with lower ΔV requirements.

The required use of drop tanks on the 14 missions that exceed the propellant capacity of the nominal payload vehicle add approximately 210 million dollars to the program cost of this vehicle. It is therefore Grumman's recommendation that a totally space based OTV be a full sized vehicle capable of capturing the forecast mission model without the benefit of auxiliary tankage.

The worst case mission presently in our mission model is the delivery of 17,500 lb to GEO in a single stage insertion, single perigee burn operating mode. Missions to be added to the model requiring a greater capacity than the above missions may or may not affect the sizing of the OTV, depending on the requirements of the mission and on the number of missions required. Therefore our recommendation on OTV sizing may change as deletions, additions, and modifications are made to the NASA mission model.

STUDY RESULTS: LARGE vs SMALL OTV

- LARGE OTV (V6 = 4-LO_2/LH_2) DESIGNED FOR 14K UP & BACK MANNED MISSION
  - DRY WEIGHT = 7000 LBS
  - TOTAL THRUST = 18000 LBS
  - I_{SP} = 480 SEC
- SMALL OTV (V20 = 11-LO_2/LH_2) DESIGNED FOR "NOMINAL" CARGO DELIVERY (13,200 LB USEFUL AT GEO) WITH PERIGEE KICK DELIVERY
  - DRY WEIGHT = 5725 LBS
  - TOTAL THRUST = 9000 LBS
  - I_{SP} = 480 SEC
- COMPARISON OF VEHICLES IN CARGO DELIVERY ROLE
  - SPACE BASED
  - PERIGEE KICK OPERATIONS

VEHICLE	# OF OTV FLTS	# OF STS FLTS
4-LO_2/LH_2	37	85
11-LO_2/LH_2*	41	85

*REQUIRES EXTERNAL TANKS FOR 14 MISSIONS

- CONCLUSIONS
  - FEWER OTV FLTS PROVIDE SMALL ENGINE LIFE ADVANTAGE TO LARGER VEHICLE
  - IF EXTERNAL TANKS MUST BE DROP TANKS, LARGE VEHICLE SAVES ~ \$210M OVER 6 YEARS

GRUMMAN

0283-002PP

A similar study to the large/small OTV tradeoff study was performed on space-based vs ground-based OTV's. The space-based OTV's have lightweight "Gossomer" structure incompatible with fully tanked OTV launches in the orbiter cargo bay. Also the Gossomer space-based OTV's do not have the upgraded insulation/purge systems required for ground-based operations. The manifesting efficiency of four space-based-only vehicles are compared to their ground-based counterparts.

Both the space-based and the ground-based vehicles are run in a space-based operating scenario in this tradeoff. The ground based vehicle carry approximately a 10% weight penalty over the space based vehicles. Over the 6 year mission model studied, the 10% weight penalty of the ground based vehicles manifested into less than 2 additional shuttle flights in all cases.

Over a six year period the addition of two extra STS flights is relatively insignificant. The cost savings over 6 years is on the order of \$160M. However, a space based only AOTV cannot efficiently service military orbits (e.g., $i = 98^{\circ}$, $i = 62^{\circ}$, and polar). This would probably cause the military to develop its own ground based AOTV, with costs on the order of \$800M. It seems more economic for the US to have one AOTV, for both NASA and the military. Thus, the weight savings gained by using light Gossomer structure is not warranted.

STUDY RESULTS: STRONG vs "GOSSOMER" SB OTV

- ISSUE: SHOULD SPACE BASED OTV BE STRONG ENOUGH TO OPERATE AS GROUND BASED, OR, SHOULD SB OTV BE MIN WEIGHT (FROM MIN STRENGTH) AND USE LESS PROPELLANT?
- STUDY COMPARED 4 PAIRS OF OTV, EACH PAIR USED SAME PROPELLANT
  - — GB OTV USED IN SB MODE
  - — SB OTV USED IN SB MODE
  - — ALL VEHICLES FLEW PERIGEE KICK OPERATIONS MODE

VEHICLE	Δ DRY WT	# OF STS FLTS	Δ # OF STS FLTS	
10-LO_2/LH_2	+575 LB	86	+1	INSIGNIFICANT OVER 6 YEARS
11-LO_2/LH_2		85		
3-LO_2/MMH	+575 LB	94	+1	
4-LO_2/MMH		93		
7-N_2O_4/MMH	+550 LB	98	+2	
8-N_2O_4/MMH		96		
6-N_2F_4/N_2H_4	+580 LB	91	-1	
7-N_2F_4/N_2H_4		92		

- CONCLUSIONS
  - — SERVICING MILITARY PAYLOADS IN POLAR ORBITS (GB FROM VAFB) WITH NASA OTV IS MORE IMPORTANT THEN THE SMALL NUMBER OF STS FLIGHTS SAVED BY SUPER LIGHT STRUCTURE ON OTV

GRUMMAN

0283-003PP

As summarized in the figure, based on performance considerations alone, LO_2/LH_2 is the best choice for an advanced OTV propellant for ground based OTV's operating in perigee kick mode (with single stage insertion of the 14 payloads which require single stage delivery). It allows the delivery of all but two of the 1995-2000 Grumman Mission Model payloads. All storable propellants failed to deliver at least 12 payloads in ground based mode.

However, it is not necessary that payloads without their own propulsion be delivered to GEO via single stage insertion. They can be attached to an apogee kick stage and delivered to a transfer orbit via a perigee kick AOTV.

In a "modified perigee kick operating mode", it was assumed that those payloads which are incapable of propulsion must be fitted with an AKS (apogee kick stage) in order to be compatible with perigee kick delivery. After running the program in this mode, it was found that there was a maximum deviation of 4 STS flights between the worst and best performing propellants. Over the 6 years of the mission model, 1995-2000, 4 shuttle flights are relatively insignificant. Therefore, based on other considerations such as technology development cost, military desire for on demand launch, and OMV propellant commonality, we recommend Nitrogentetraoxide/monomethylhydrazine as the propellant for an advanced ground based AOTV. The results of the manifesting analysis are summarized in the figure.

STUDY RESULTS: OTV PROPELLANT SELECTION

- COMPARED 2 SETS OF OTVs WITH 4 DIFFERENT PROPELLANTS IN BOTH GROUND & SPACE BASED MODES
  - EACH SET CONTAINED 4 VEHICLES, OPTIMIZED FOR THAT BASING MODE, & THE PROPELLANT/ ENGINES THAT WE HAD DATA ON

GROUND BASED VEHICLES (BASING MODE = 1):

	PERIGEE KICK OPERATION MODE			MODIFIED PK OPERATION MODE: ALL PAYLOADS FLY PERIGEE KICK		
	# OF PAYLOADS FLOWN					
VEHICLE	"GOOD FIT" OUTPUT*	ADJUSTED OUTPUT	# OF PAYLOADS THAT DO NOT FLY	# OF PAYLOADS FLOWN	# OF STS FLTS	Δ STS FLTS
10-LO_2/LH_2	162	174	2	176	93	+3
3-LO_2/MMH	161	164	12	176	90	0
7-N_2O_4/MMH	161	162	14	176	90	0
6-N_2F_4/MMH	160	163	13	176	89	-1

*PROGRAM LOGIC, NOT STS OR OTV CAPACITY, PREVENTED 17 TO 19 PAYLOADS FROM BEING MANIFESTED

- CONCLUSIONS
  - ALL GROUND BASE "STORABLES" OUTPERFORMED LO_2/LH_2
  - N_2O_4/MMH IS PREFERRED
    - LOWEST DEVELOPMENT COST
    - LEAST EXPENSIVE WAY OF SATISFYING MILITARY DESIRE FOR ON-DEMAND LAUNCH
    - SAME PROPELLANTS ON OMV

GRUMMAN

0283-004(1/3)PP

In a fully space based mode, when delivering payloads using the "perigee kick" operating mode, the LO_2/LH_2 AOTV's performed significantly better than the storable propellant AOTVs. It is shown that LO_2/LH_2 has an average one to two flight/year advantage over the 6 year mission model when compared to the other propellant combinations under consideration. However, other operational considerations as well as economic constraints favor the storable propellants.

Further analysis is required to determine the economic impact of the required engine technology programs, and the cost of propellant storage at the space station. The cost of the additional STS flights required by the storable propellant combinations must also be defined in order to predict the LCC of all candidate AOTV's.

OTV PROPELLANT SELECTION

SPACE BASED OTVs (BASING MODE = 4):

	VEHICLE	"GOOD FIT" OUTPUT, PERIGEE KICK MODE: # OF PAYLOADS FLOWN	# OF STS FLTS	Δ STS FLTS	
GB	10-LO_2/LH_2	176	86 • 14/YEAR	0	
	3-LO_2/MMH	176	95	+9	1-1/2/YR
	7-N_2O_4/MMH	176	98 • 16/YEAR	+12	2/YR
	6-N_2F_4/N_2H_4	176	92	+6	1/YR
SB	11-LO_2/LH_2	176	85	0	
	4-LO_2/MMH	176	93	+8	1-1/3/YR
	8-N_2O_4/MMH	176	96	+11	2/YR
	7-N_2F_4/N_2H_4	176	92	+7	1/YR

- CONCLUSIONS
  - SIGNIFICANT ADVANTAGE OF LO_2/LH_2 OVER N_2O_4/MMH
  - NOT SO SIGNIFICANT ADVANTAGE OF LO_2/LH_2 OVER N_2F_4/N_2H_4
  - FURTHER ANALYSIS IS REQUIRED
    - NEW ENGINE DEVELOPMENT COST
    - COST OF ADDITIONAL STS FLIGHTS
    - COST OF STORING & TRANSFERING CRYO HYDROGEN @ SS
    - COST OF STORING & TRANSFERING CYRO OXYGEN @ SS
    - COST OF STORING & TRANSFERING CYRO N_2F_4 @ SS

SPACE BASED OTVs (CONT):

- SWAG AT ABOVE ANALYSIS PRODUCED FOLLOWING RANKING, WITH TOTAL COSTS FOR 8 YEARS OF FLIGHT + DDT & E, OF $1B
  1. N_2O_4/MMH ⟶ LOWEST COST
  2. LO_2/MMH +6%
  3. LO_2/LH_2 +10%
  4. N_2F_4 +19%
- CONCLUSIONS
  - ECONOMIC JUSTIFICATION REQUIRES MORE DETAILED STUDY OF SPACE BASED OPERATIONS
    - PROPELLANT STORAGE & HANDLING
  - AT THIS TIME, N_2O_4/MMH IS PREFERRED

GROUND BASED & SPACE BASED PROPELLANT SELECTION:

- AT THIS TIME, N_2O_4/MMH IS RECOMMENDED FOR CARGO OTV
  - LOWEST UP FRONT COSTS
    - DEVELOPMENT
    - EARLY YEARS OF OPERATION, WHICH WILL INCLUDE MANY GROUND BASED MISSIONS
  - MOST COMPATIBLE WITH MILITARY NEEDS

QUALIFICATION:

- MANNED MISSIONS WERE NOT CONSIDERED IN THIS MANIFESTING STUDY
- OTHER WORK HAS SHOWN A SAVINGS OF 1 STS FLT/MANNED MISSION WITH A SMALL CREW CAPSULE ("BARE BONES") AND SMALL, HIGH I_{sp} LO_2/LH_2 ENGINES

GRUMMAN

An examination of space station technology payoffs was conducted. A fully ground based OTV with ground based operations was considered the baseline condition. The two alternate scenarios evaluated were a ground based OTV with space station payload manifesting, and a ground based OTV with space station payload and propellant manifesting. The breakdown of STS flights saved by the alternate operating scenarios is summarized. After considering the cost of payload manifesting at space station, we have concluded that they are minimal (since comparable SS infrastructure exists to handle SS payloads). Therefore AOTV payload manifesting should be implemented at the space station. A tentative cost/benefit analysis on propellant storage at the space station has yielded results that lead to the conclusion that the benefits associated with the introduction of propellant manifesting in the cases of N_2O_4/MMH, N_2F_4/N_2H_4 and LO_2/MMH OTV's outweigh the costs. Based on this preliminary analysis, we recommend that space station propellant manifesting be implemented for N_2O_4/MMH, N_2F_4/N_2H_4, LO_2/MMH AOTV's.

STUDY RESULTS: SPACE STATION TECHNOLOGY

- ENGINE RELATED TOPICS
  - — ADVANCED ENGINE NOZZLE STRENGTH IS ADEQUATE
  - — FOR ADVANCED LO_2/LH_2 ENGINES
    - RECOMMENDED TOTAL VEHICLE THRUST = 8000 LB TO 13000 LB FOR "NOMINAL" PAYLOAD CARGO OTV
    - RECOMMEND 4 3000 LB THRUST ENGINES FOR "NOMINAL" PAYLOAD CARGO OTV
- VEHICLE CONFIGURATIONS
  - — DEVELOPED MANY NEW VEHICLES FOR USE IN MANIFESTING STUDY
    - MIN WEIGHT SPACE BASED
    - MIN LENGTH GROUND BASED
  - — GB VEHICLES WEIGH ~ 570 LB (10%) MORE THAN SB OTVs
- MANIFESTING STUDY CONCLUSIONS
  - — SPACE BASED OTV SHOULD BE "FULL SIZE"
  - — SPACE BASED OTV SHOULD BE CAPABLE OF OPERATING IN A GROUND BASED MODE
  - — AT THIS TIME, N_2O_4/MMH IS THE PREFERRED PROPELLANT FOR THE NEXT CARGO CARRYING OTV
    - INTEGRATING MANNED MISSIONS INTO THIS CONCLUSION REQUIRES FURTHER STUDY
    - ADDITIONAL STUDY OF SPACE BASE OPERATIONS IS NEEDED FOR AN ECONOMIC JUSTIFICATION OF OTV PROPELLANT SELECTION
  - — SPACE STATION PAYLOAD MANIFESTING IS RECOMMENDED
  - — AT THIS TIME, SS PROPELLANT MANIFESTING IS RECOMMENDED FOR ALL PROPELLANTS EXCEPT LO_2/LH_2

- COMPARED AOTVs USING DIFFERENT BASING MODES TO DETERMINE THE REDUCTION IN TOTAL STS FLIGHTS FOR DIFFERENT LEVELS OF SPACE STATION ACTIVITY
- ALL RESULTS USED MODIFIED PERIGEE KICK OPERATING MODE
  - — 15 PAYLOADS ORIGINALLY SCHEDULED FOR SINGLE STAGE DELIVERY TO GEO WERE DELIVERED VIA PERIGEE KICK
- BASING MODE CODE:
  - 1 = GROUND BASED AOTV
  - 2 = GROUND BASED + SPACE STATION PAYLOAD MANIFESTING
  - 3 = GROUND MAINTAINED AOTV + SPACE STATION PAYLOAD & PROPELLANT MANIFESTING

BASING MODE	VEHICLES	# OF PAYLOADS FLOWN	# OF STS FLIGHTS	Δ STS FLTS	STS FLTS SAVED BY SS PAYLOAD MANIF.	STS FLTS SAVED BY SS PROPELLANT MANIF.
1	10-LO_2/LH_2	176	93	0		
2	10-LO_2/LH_2	176	90	-3	3	4
3	10-LO_2/LH_2	176	86	-7		
1	3-LO_2/MMH	176	90	0		
2	3-LO_2/MMH	176	88	-2	2	5
3	3-LO_2/MMH	176	83	-7		
1	7-N_2O_4/MMH	176	90	0		
2	7-N_2O_4/MMH	176	89	-1	1	5
3	7-N_2O_4/MMH	176	84	-6		
1	6-N_2F_4/N_2H_4	176	89	0		
2	6-N_2F_4/N_2H_4	176	87	-2	2	4
3	6-N_2F_4/N_2H_4	176	83	-6		

- SS PAYLOAD MANIFESTING
  - — MOST OF REQ'D INFRASTRUCTURE EXISTS TO HANDLE SS PAYLOADS
    - SMALL COST ASSOCIATED WITH ADDING AOTV PAYLOAD HANDLING
    - ANY SAVING OF STS FLIGHTS WILL YIELD A PROFIT
  - — RECOMMENDATION: INCORPORATE SPACE STATION PAYLOAD MANIFESTING FOR GROUND BASED AOTVs
- SS PROPELLANT MANIFESTING
  - — BENEFIT A STRONG FUNCTION OF COST ADDED TO SS TO STORE & DISTRIBUTE PROPELLANTS

PROPELLANT	SWAG @ SS DDT & E	STS FLT BENEFIT	ESTIMATED NET GAIN (+) OR LOSS (-)
LO_2/LH_2	-$700M	+$330M	-$370M
LO_2/MMH	-$120M	+$422M	+$302M
N_2O_4/MMH	-$30M	+$422M	+$392M
N_2F_4/N_2H_4	-$160M	+$335M	+$175M

  - — RECOMMENDATION: AT THIS TIME, INCORPORATE SPACE STATION PROPELLANT MANIFESTING FOR ALL PROPELLANTS EXCEPT LO_2/LH_2

3.2.4 Low L/D AOTV Concerns and Issues

Current NASA studies also include AOTVs in the L/D range of 0 to 0.75. Candidate configurations include several that employ a fabric surface and one that is inflatible, a ballute. We have performed a cursory review of these configurations and have listed here some of the main aerothermodynamic issues that we feel need increased attention.

RE-ENTRY SYSTEMS OPERATIONS

WHAT ARE SOME MAIN ISSUES FOR LOW L/D

- MACRO MODELING OF POROUS CLOTH AEROTHERMAL RESPONSE IS REQUIRED

  POROSITY, ROUGHNESS, EMITTANCE, GEOMETRY INDUCED $\partial p/\partial x$, VIABLE TESTS, FLEXING AT VERY LOW AND VERY HIGH TEMPERATURES, LOW CATALYTIC SURFACE

- AOTV-PAYLOAD/SEPARATED FLOW REGION

  - GEOMETRY
  - HEAT TRANSFER RATES
  - FLIGHT/TEST SIMULITUDE

Some examples of the effects of geometry on heat transfer enhancement are illustrated here. On the nearly flat faced model, even at zero angle of attack, the local heat transfer at the model perimeter is nearly twice that at the model center. Tailoring of the corner geometry can reduce local heat transfer.

The chart on the right illustrates effects on heat transfer enhancement of local surface irregularities. It is expected that if drag modulation for control purposes is accomplished by varying ballute inflation pressure, surface irregularities may result in serious local heat transfer uncertainties/enhancement.

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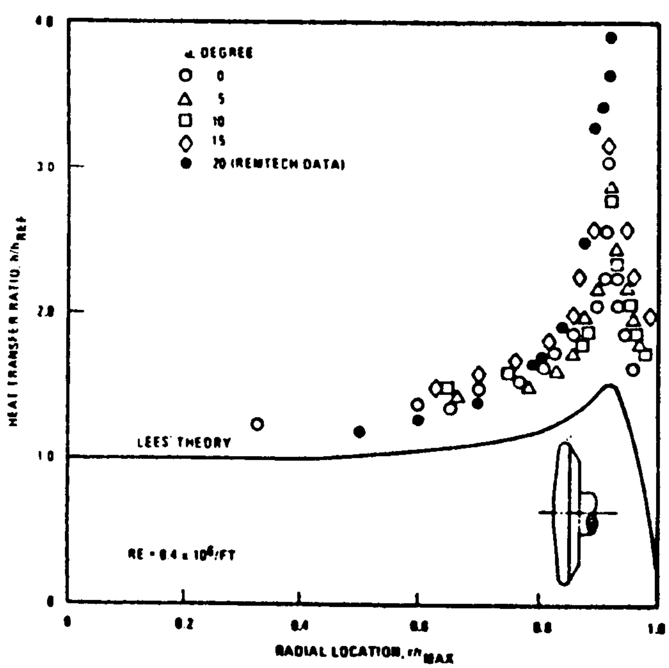
BALLUTE & BRAKE HEATING CONCERNS

CURVATURE EFFECTS – WHAT IS GEOMETRY ?

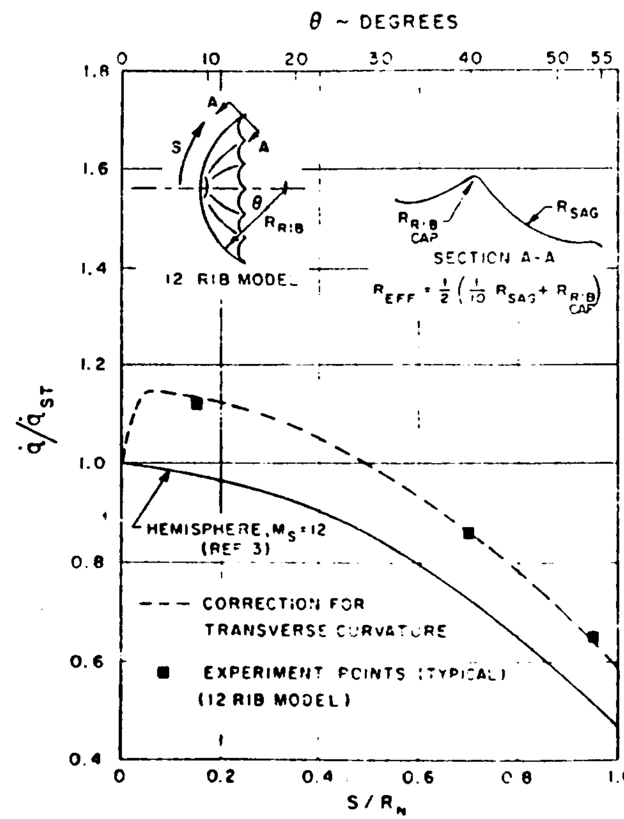
EFFECT OF CORNER RADIUS AND ANGLE OF ATTACK



Radial variation of heat transfer at low Reynolds number

SHIH AND GRAY
AIAA PAPER # 84-0309

EFFECT OF RIB PROTRUSION



Comparison of the theoretical correction for the effect of rib protrusion with experiment

GOLDBERG AND STANKEVICS
JSR FEB 1962

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One of the current heat protection concepts proposed for the Ballute and Brakes is a porous cloth backed by a layer of low density insulation. The porous cloth/insulation layer in the presence of local pressure gradients can result in significant local surface and internal heat transfer amplification. This chart outlines our concern and a technique that we employed to solve this problem on a non-reusable design.

RE-ENTRY SYSTEMS OPERATIONS

BALLUTE & BRAKE HEATING CONCERNS

POROUS CLOTH EFFECTS

BALLUTE DRAG MODULATION REQUIREMENT GENERATES VARIABLE SOFT GEOMETRY

↓

LOCAL GEOMETRY DRIVES LOCAL PRESSURE GRADIENT $\partial p/\partial x$

↓

LOCAL PRESSURE GRADIENT ON A POROUS CLOTH DRIVES LOCAL SUCTION AND BOUNDARY LAYER THINNING

↓

RESULTS IN SIGNIFICANT LOCAL HEAT TRANSFER AMPLIFICATION

↓

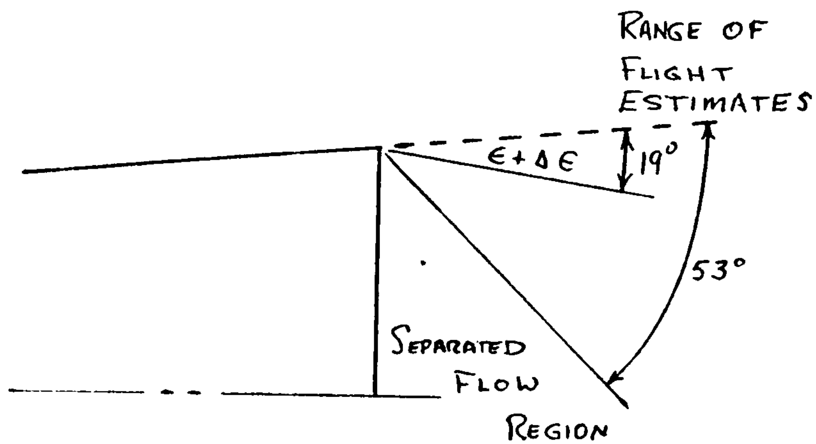
WE SOLVED PROBLEM FOR NON-REUSABLE SYSTEM BY SEALING CLOTH WITH RTV 560 - UNACCEPTABLY HEAVY

A major problem/issue for all AOTVs is the magnitude of the local heat transfer in the base flow region. Details of the geometry of this region and the expected heat transfer rates in the separated flow region are described in other sections of this report. The aerothermodynamic similitude problem is described here, underlining the need for careful/expert ground test analysis/planning and the desirability of a flight test experiment.

RE-ENTRY SYSTEMS OPERATIONS

AEROTHERMODYNAMIC SIMILITUDE

FOR V_∞ = 32000 FT/SEC
γ = 1.4
T_W = 1500°R
SHARP CONE PROPERTIES



FLIGHT	M_∞ = 30 γ = REAL
FLIGHT TEST	M_∞ = 20 TO 25 γ = REAL
GROUND TEST	M_∞ = 6 TO 10 γ = CONTROLLED

ϵ – DOMINATED BY $\theta_{C_{EFF}}$ AND γ

$\Delta\epsilon$ – AFFECTED BY $\theta_{C_{EFF}}$, L, H AND γ

γ – AFFECTED BY M_∞, H, CONFIGURATION

3.3 Propulsion S/S

3.3.1 Mid L/D Base Flow Wake Closure and Separted Flow Heat Transfer

Engine choices being considered for the Mid L/D AOTV range from one large 15 to 20K lbs thrust engine to six small 2500 to 4000 lb engines. If the engine nozzle(s) is allowed to extend beyond the end of the vehicle, it is expected that the AOTV dry weight will be less than if it is necessary to extend the AOTV fuselage to cover most or all of the nozzle(s). Based on previous experience and the Space Shuttle flight result, it is expected that emersion of the engine nozzles in the separated flow region and the relatively benign heating will prove acceptable, while direct flow and shock impingement from that flow turning the corner at the aft end of the AOTV will prove unacceptable based on the exceptionally high heating rates expected.

Evaluation of hypersonic heat transfer in the base region of the AOTV is divided into two primary activities. The first is that of defining the geometry of the relatively quiescent separated flow region; the second is that of predicting the hypersonic heat transfer rates in this separated flow region.

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HYPERSONIC FLOW BASE HEAT TRANSFER

C-3

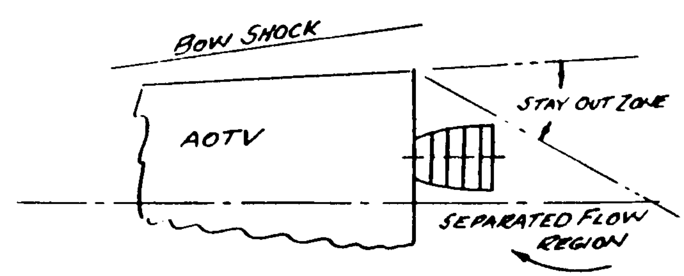
BACKROUND

- STRATEGIC R/V HERITAGE - LAMINAR & TURBULENT FLOW DESIGN ALGORITHM AVAILABLE FOR SHARP AND BLUNT BODIES
- MERCURY AND APOLLO GROUND TEST - EXTENDED DATA BASE INTO LOWER REYNOLDS NUMBER REGION

PHASE II TASK

- ASSEMBLE SHUTTLE ORBITER HYPERSONIC BASE HEAT TRANSFER FLIGHT TEST DATA AND COMPARE TO ABOVE - NEED MSFC/REMTECH DATA DUMP
- SELECT A DESIGN APPROACH FOR THIS STUDY
- EVALUATE HYPERSONIC SEPARATED FLOW BASE HEAT TRANSFER TO MID L/D AOTV OVER A RANGE OF REPRESENTATIVE MINIMUM ALTITUDES

- o GEOMETRY OF WAKE EXPANSION - STAY OUT ZONES
  - BASE PRESSURE
  - PRANDTL MEYER EXPANSION
- o SEPARATED FLOW BASE HEAT TRANSFER
  - FLAT BASE
  - WITH ENGINE NOZZLE EXTENDED INTO SEPARATED FLOW REGION
  - WITH BODY FLAP INDUCED SHOCK IMPINGEMENT



Prior to selecting a narrow class of biconic mid L/D AOTVs, estimates were made of the impact of direct flow impingement on engine nozzles protruding from the AOTV aft end. Results are outlined in the Figure.

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MID L/D AOTV NOZZLE LOCATION ISSUES

- REATTACHED HYPERSONIC FLOW NOZZLES EXPECTED TO PRODUCE HEAT TRANSFER RATES OF 0.4 TO 0.7 OF q reference
  - WOULD DESTROY ENGINE NOZZLE

- SEPARATED FLOW EXPECTED TO PRODUCE HEAT TRANSFER RATES < 0.01 q reference

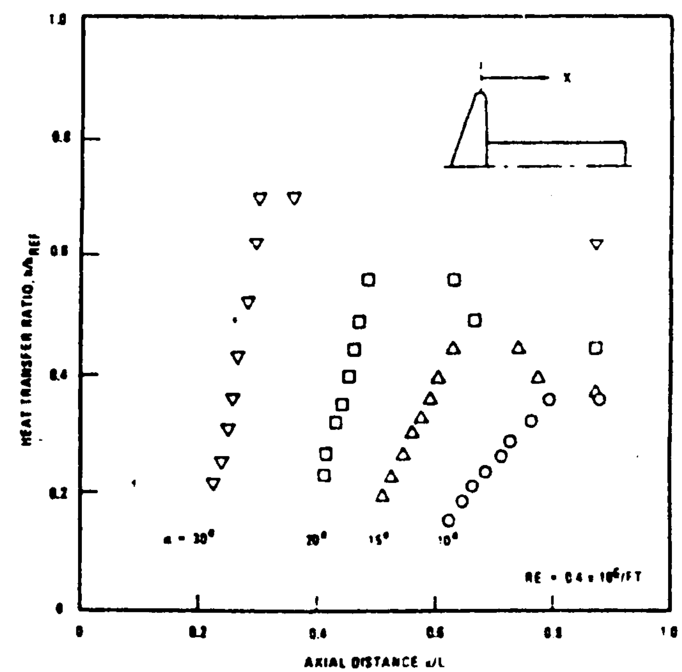

Figure 19. Afterbody heat transfer distribution (Option 3).

- CONCLUDE : HYPERSONIC ATTACHED FLOW TO PROTRUDING NOZZLES IS UNACCEPTABLE

First, to define the geometry of the separated flow region it is necessary to determine the angle of the flow as it turns the AOTV aft corner. The flow turning angle is dependent on numerous parameters, including the pressure in the AOTV based flow region and the local specific heat ratio of the flow as it turns into the base region. Correlations of ground and high Mach number flight test base pressure data in separated laminar flow regions behind both blunt and sharp cone type vehicles are available. The sharp correlation was felt to be more applicable for the mid L/D biconic type AOTV and was employed to estimate the range of base pressures expected.

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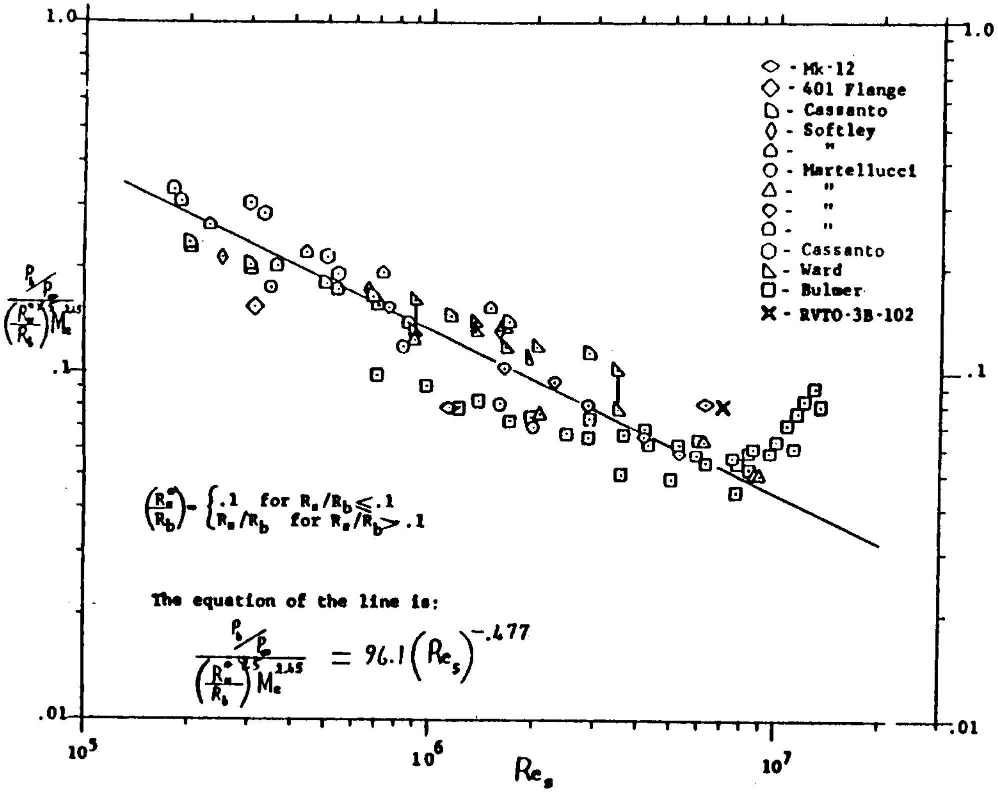
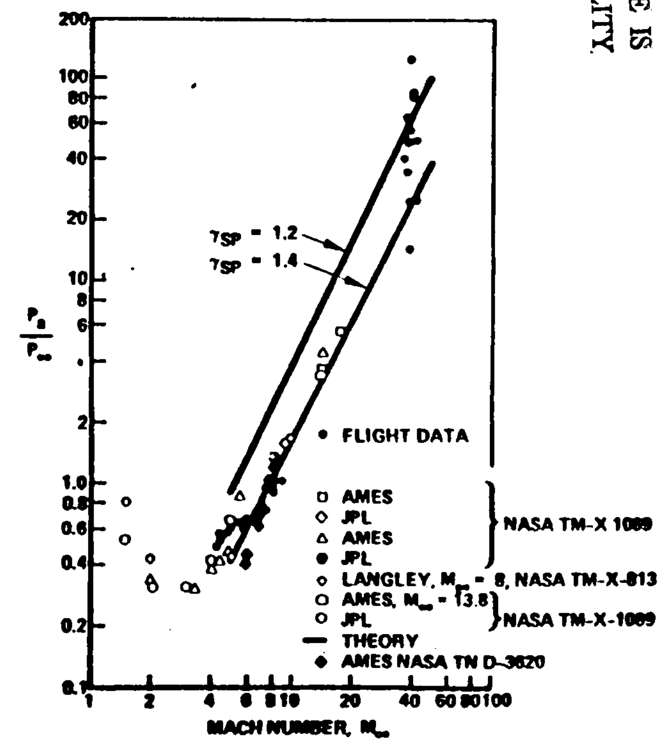
CORRELATION OF FLIGHT MEASUREMENTS OF BASE PRESSURES ON CONICAL SHAPED BODIES – LAMINAR FLOW – KAWECKI, JSR MAY 1977



CORRELATION OF FLIGHT AND WIND TUNNEL MEASUREMENTS OF AFTERBODY PRESSURES ON APOLLO-SHAPED BODIES

– LAMINAR FLOW



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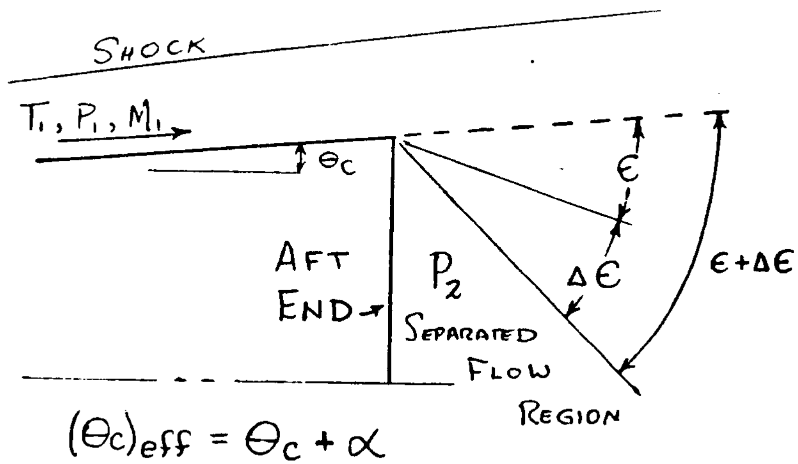
Prior to selecting a narrow class of biconic mid L/DC AOTVs, estimates were made of the wake flow expansion angles for a range of AOTVs using an assumed specific heat ratio of $\gamma = 1.4$, and the procedure outlinedon Page . Results are summarized on the following page.

RE-ENTRY SYSTEMS OPERATIONS

SUMMARY OF CALCULATED EXPANSION ANGLES BASED ON $\gamma = 1.4$ AND SHARP CONE PROPERTIES AND $T_W = 1500R$, $V_\infty = 32,000$ ft/sec



ϵ = WAKE EXPANSION ANGLE FROM SURFACE DIRECTION

$\Delta\epsilon$ = ANGLE INCREMENT CORRESPONDING TO SHEAR LAYER WIDTH

ALT Kft	$(\theta_c)_{eff}$ deg	L ft	BASE PRES. P_2/P_0	ε deg	$\Delta\varepsilon$ deg	$\varepsilon + \Delta\varepsilon$ deg	$\varepsilon - (\theta_c)_{eff}$ deg
190	10	10	1.33	10.1	18.0	28.1	0.1
		60	0.565	11.9	6.9	18.8	1.9
	20	10	0.272	29.3	6.1	35.4	9.3
		60	0.116	31.5	2.5	34.0	11.5
240	10	10	3.37	8.4	45	53.4	-1.6
		60	1.43	10.4	18	28.4	0.4
	20	10	0.686	27.3	16	43.3	7.3
		60	0.291	29.7	6.5	36.2	9.7

Conclusions drawn from review of the results summarized on the previous page are listed.

RE-ENTRY SYSTEMS OPERATIONS

MID L/D AOTV BASE SEPARATED FLOW REGION CONCLUSIONS

ϵ – DOMINATED BY $\theta_{C_{EFF}}$ AND γ

$\Delta\epsilon$ – AFFECTED BY $\theta_{C_{EFF}}$, L, H AND γ

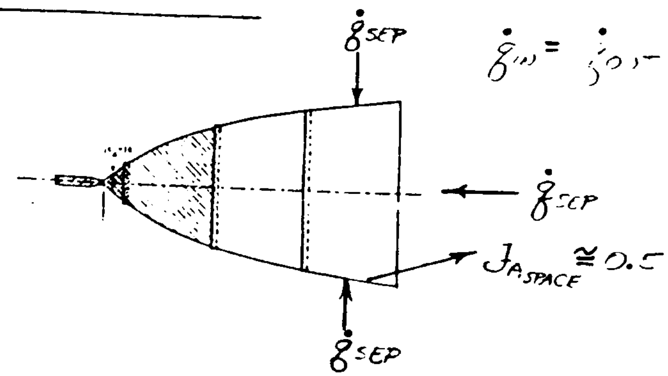
γ – AFFECTED BY M_∞, H, CONFIGURATION

The AOTV configuration/engine nozzle geometry is selected to avoid direct flow impingement on the nozzles during hypersonic heating. Estimates were made of the magnitude of the separated flow heat transfer.

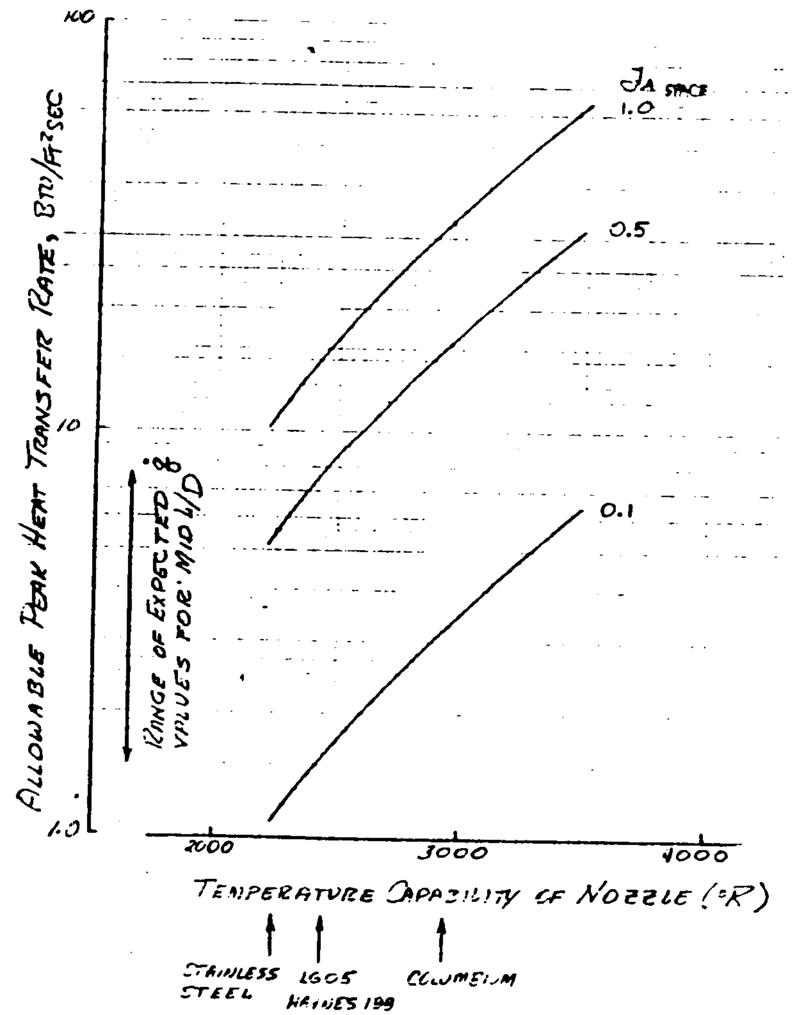
RE-ENTRY SYSTEMS OPERATIONS

MID L/D AOTV NOZZLE LOCATION ISSUES - HYPERSONIC FLIGHT

AOTV SHELL



- GOOD VIEW OF SPACE ESSENTIAL
- $\dot{q}_{SEP}$ ESTIMATED TO BE $0.01 \dot{q}_{REF}$
- $\dot{q}_{SEP}$ EXPECTED TO DROP RAPIDLY INSIDE NOZZLE
- $\dot{q}_{REF}$ EXPECTED TO VARY FROM MAX OF 150 TO 800 BTU/FT^2 SEC
- CONCLUDE: SEPARATED FLOW HEAT TRANSFER LOOKS TOLERABLE, TEMPERATURE CAPABILITY > 2000°F IS REQUIRED



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Real gas flow field calculations were performed to determine local conditions at the boundary layer edge. Flow turning angle, ϵ, due to the expansion into the wake area was estimated using a technique suggested by Nestler where the local properties in the separated flow region were computed by performing a Prandtle-Meyer expansion from the local boundary layer edge conditions to the base pressure.

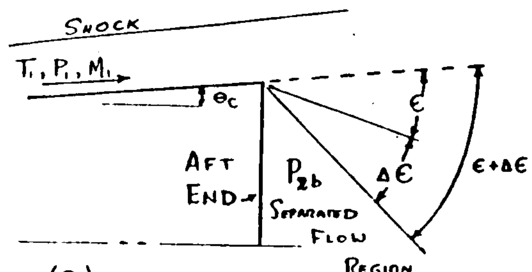
The angle increment, $\Delta\epsilon$, corresponding to the shear layer width was computed using the Chapman shear layer analysis where

$$\Delta\epsilon = \frac{4.28 + 2.57\ (Tw/T1 - 1) + 0.48\ \left(\frac{\gamma - 1}{2}\right) M_1^{\,2}}{\left(Res_1/10^6\right)^{1/2}}$$

This $\Delta\epsilon$ corresponds to the u = 0 location in the shear layer profile, selected for conservatism since there is little data at angle of attack. These calculated values of $\epsilon + \Delta\epsilon$ are summarized in the next two charts for mid L/D AOTVs representative of the shortest and the longest vehicles, angles of attack up to 20°, and altitudes of 200 and 300 kft.

Table 2. Summary of Calculated Expansion Angles Based on Real Gas γ and Computed Local Properties
$T_W = 1500°R$ $M_\infty = 30$

Alt Kft	$(\theta_c)_{eff}$ deg	L ft	P_B/P_C	ϵ deg	$\Delta\epsilon$ Deg	$\epsilon + \Delta\epsilon$ Deg	$\epsilon - (\theta_c)_{eff}$ Deg
200	6	34	5.17	3.5	11.5	15.0	-2.5
		51	4.08	5.3	9.0	14.3	-0.7
	11	34	3.75	8.6	8.3	16.9	-2.4
		51	2.7	10.7	6.3	17.0	-0.3
	16	34	2.29	14.9	5.2	20.1	-1.1
		51	1.61	18.1	3.8	21.9	+2.1
	21	34	1.27	23.9	3.4	27.3	+2.9
		51	0.79	28.4	2.2	30.6	+7.4
300	21	34	11.4	14.0	34	48	-7.0



$(\theta_c)_{eff} = \theta_c + \alpha$

ϵ = WAKE EXPANSION ANGLE FROM SURFACE DIRECTION

$\Delta\epsilon$ = ANGLE INCREMENT CORRESPONDING TO SHEAR LAYER WIDTH

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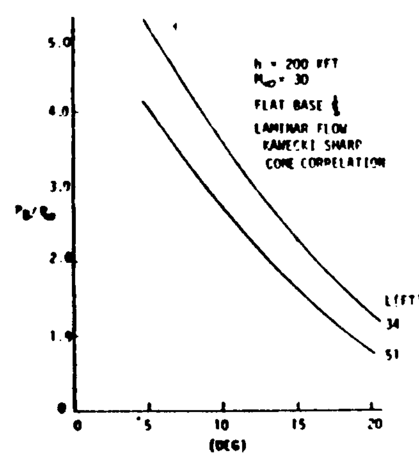
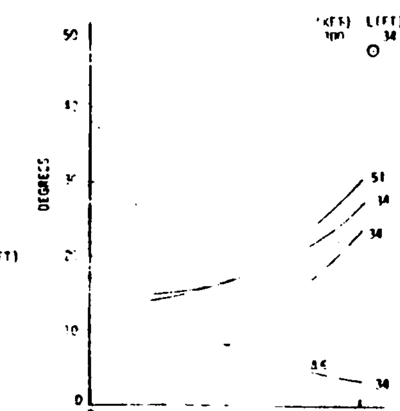




Substantial uncertainty exists regarding predicting ϵ and $\Delta\epsilon$ and uncertainty is greater at higher altitudes (local BL edge conds). $\Delta\epsilon$ is proportional to $1/(Res)^{1/2}$ and hence, highly altitude dependent.

The 34 ft vehicle experienced an increase in $\Delta\epsilon$ from 3.5 to 34^o for h going from 200 to 300kft and a decrease in ϵ from 24^o to 14^o, hence $(\epsilon + \Delta\epsilon)$ varying from 27 to 48^o (200 to 300 kft) for $\alpha = 20^o$. Now, $\dot{q}300 = 0.1\ \dot{q}200$, but since with impingement, the $\dot{q}$ increases by X10 to X50, we must avoid full impingement even at 300 kft.

A conservative design approach would use $\epsilon + \Delta\epsilon$ @ 300 kft. This would result in extensive - almost total protection ($\Delta\epsilon \cong 83^o$ for $\alpha = 10^o$, L = 34 ft) of the nozzle. Actual detailed analysis may produce an answer of allowable bound closer to ϵ at 200 kft. So - performance payoff exists for decreasing uncertainty regarding base heat transfer. This would include analyses, detailed calibration of methodology, low Reynolds tests.

STAY OUT ZONES CONCLUSIONS

- SUBSTANTIAL UNCERTAINTY EXISTS RE PREDICTING ϵ & $\Delta\epsilon$ AND UNCERTAINTY IS GREATER AT HIGHER ALTITUDES (LOCAL BL EDGE CONDS)

- $\Delta\epsilon$ IS $\sim 1/(RES)^{1/2}$ & HENCE, HIGHLY ALTITUDE DEPENDENT

- 34 VEHICLES EXPERIENCED AN INCREASE IN $\Delta\epsilon$ FROM 3.4 TO 34^o FOR H GOING FROM 200 TO 300K FT AND A DECREASE IN ϵ FROM 24 TO 14^o; HENCE ($\epsilon + \Delta\epsilon$) VARYING FROM 27 TO 48^o (200 TO 300K FT) FOR $\alpha = 20^o$

- SINCE $\dot{q}_{300} \cong 0.1\ \dot{q}_{200}$, BUT SINCE WITH IMPINGEMENT, THE $\dot{q}$ INCREASES BY X10 TO X50 WE MUST AVOID FULL IMPINGEMENT EVEN AT 300 K FT.

- CONSERVATIVE DESIGN APPROACH WOULD USE $\epsilon + \Delta\epsilon$ @ 300K. THIS WOULD RESULT IN EXTENSIVE - ALMOST TOTAL PROTECTION ($\Delta\epsilon = 83^o$ for $\alpha = 10^o$, $L = 34'$)

- ACTUAL DETAILED ANALYSIS MAY PRODUCE ANSWER OF ALLOWABLE BOUND CLOSER TO ϵ AT 200K FT, SAY 30^o

- SO - PERFORMANCE PAYOFF EXISTS FOR DECREASING UNCERTAINTY RE BASE HEAT TRANSFER
  - DETAILED ANALYSIS
  - CALIBRATION OF METHODOLOGY
  - LOW REYS, HIGH M_∞ TESTS

The second part of AOTV hypersonic separated base flow computations involves prediction of the heat transfer to a vehicle with a flat base, then with engine nozzle extended into the separated flow region, and finally with body trim flap present.

MID L/D BASE HEAT TRANSFER

- ESTABLISHED METHODS USED TO PREDICT SEPARATED FLOW CONVECTIVE HEAT TRANSFER TO FLAT BASE (± X2.5)
- SUBSTANTIAL UNCERTAINTY EXISTS RE P_B/P_∞ (SHARP CONE VS. BLUNT BODY CORRELATIONS)
- CREATES UNCERTAINTY IN $\dot{q}_c$
- STS DATA SUGGESTS PRESENCE OF NOZZLE IN BASE REGION PRODUCES HEAT TRANSFER AMPLIFICATION OVER FLAT BASE NUMBERS OF $\sim$ X10
- STS DATA SUGGESTS BODY FLAP DEFLECTION INDUCED SHOCK IMPINGEMENT PRODUCES ADDITIONAL HEAT TRANSFER AMPLIFICATION $\sim$ X4
- THESE AMPLIFICATION EFFECTS HAVE BEEN COMBINED AND APPLIED TO THE UNDISTRIBUTED FLAT BASE NUMBERS.

Established methods have been used to predict separated flow convective heat transfer to a flat base area. Heat transfer data in the separated flow base region of re-entry vehicles has been summarized by Nestler (1) and shown in the figure, and provides an upper bound in the low Reynolds number region to anchor the AOTV separated flow heat transfer design predictions. The early laminar and turbulent flow correlations of base Stanton number as a function of base Reynolds number were derived from early MK2 blunt re-entry vehicle flight test data (2). The base Reynolds number and Staton number are computed employing local static properties in the base region, obtained by expanding isentropically from the brake edge conditions to the base pressure. Additional pertinent data was obtained during the Mercury and Apollo program extending into the low Reynolds number range (3).

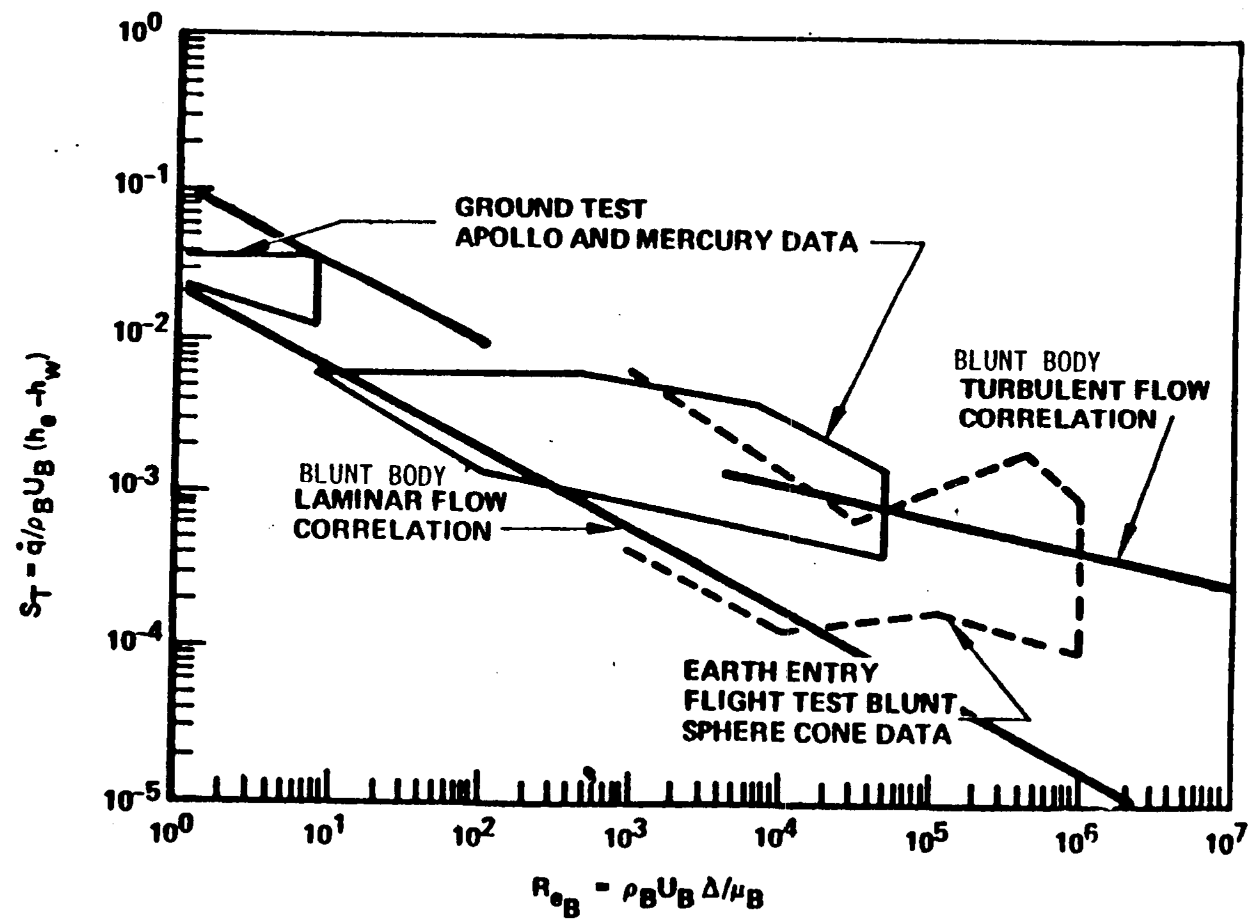
1. Nestler, D.E., and Hakett, V.F., Lifting Vehicle Heat Transfer Technology, GE-RSD-TFM8151-040, April 1965.

2. Walker, G.K., Shaw, T.W., Schmitt, G.W., and Haverly, G.C., Final Summary Report - Aerothermodynamic Analysis of Mark 2 Flight Test Data, GE-R60SD481, January 1961.

3. Lee, G. and Sundell, R.E., Apollo Afterbody Heat Transfer and Pressure with and without Ablation at M of 5.8 to 8.3, NASA TN D-3620, September 1966.

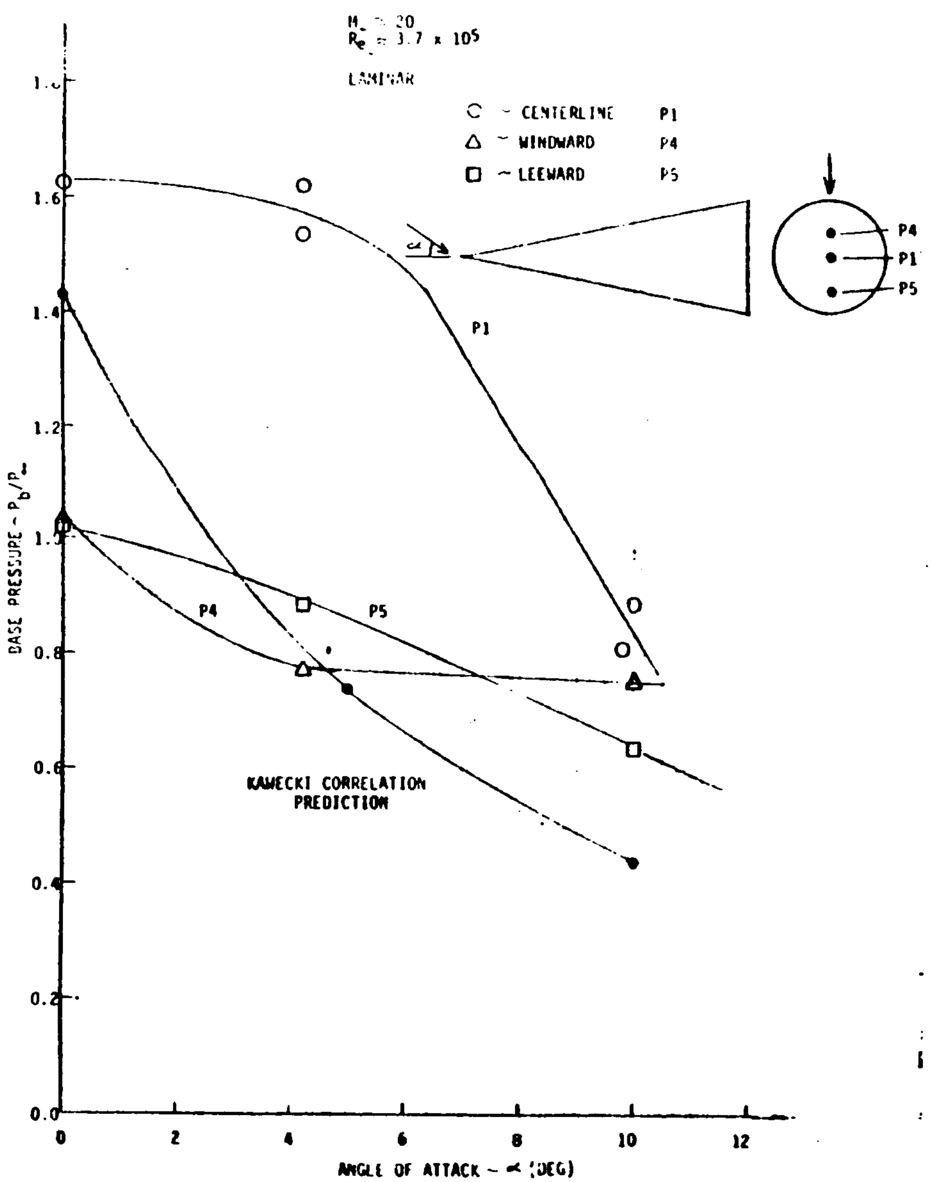
RE·ENTRY SYSTEMS OPERATIONS

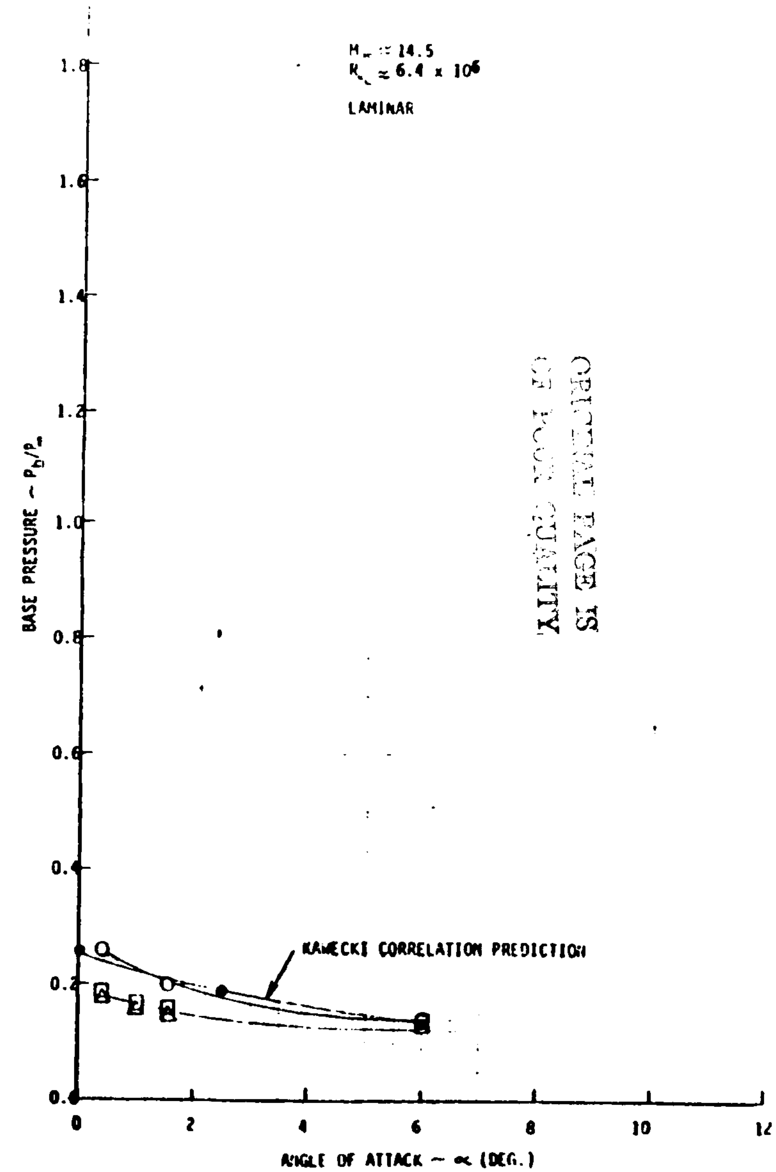
HYPERSONIC SEPARATED FLOW HEAT TRANSFER DATA BASE



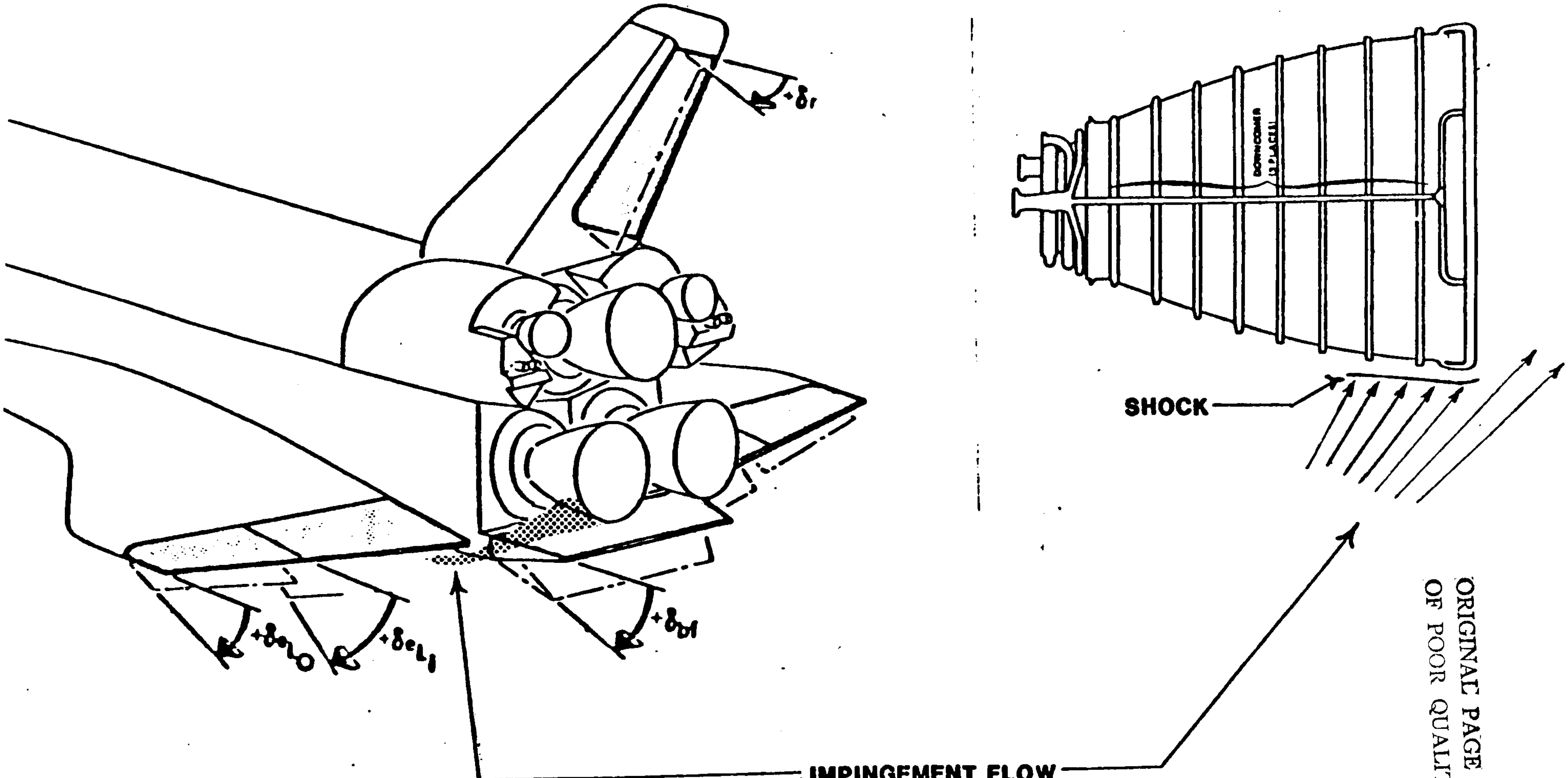
It can be seen that in the laminar flow region, an uncertainty also exists in prediction of base pressure (PB/P_∞), which creates an additional uncertainty in the base Stanton number. Review of the laminar base pressure correlation for pointed cones, Page   and the comparison at angle of attack on the next page, suggests an uncertainty of X2. This translates into an uncertainty of about X1.5 in the predicted base Stanton number.

ADDITIONAL UNCERTAINTY EXISTS ON ANGLE OF ATTACK EFFECT ON BASE PRESSURE

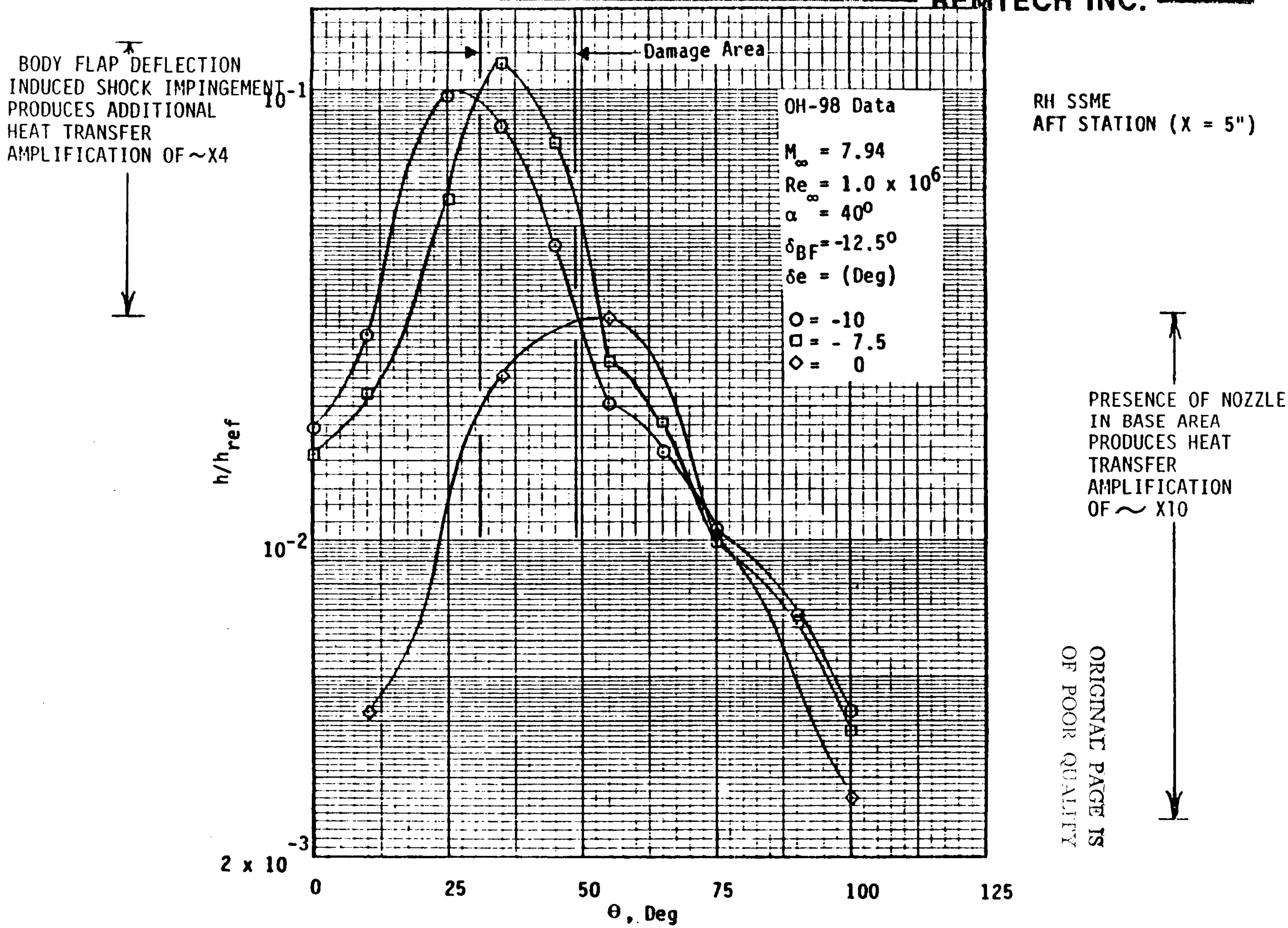




The complex geometrical relationship of body flap induced heat transfer amplification on the SSMEs due to shock impingement is illustrated for the Space Shuttle Orbiter.


+δr
+δel o
+δel i
+δbf
IMPINGEMENT FLOW
SHOCK
DOWNCOMER (3 PLACES)

Extension of the engine nozzles into the separated flow base region produces heat transfer rate amplification of X10 on the nozzles, according to the STS data provided by Foster. Additional heat transfer amplification of X4 was observed due to STS body flap deflection induced shock impingement.


BODY FLAP DEFLECTION
INDUCED SHOCK IMPINGEMENT
PRODUCES ADDITIONAL
HEAT TRANSFER
AMPLIFICATION OF ~X4
Damage Area
OH-98 Data
$M_\infty = 7.94$
$Re_\infty = 1.0 \times 10^6$
$\alpha = 40^o$
$\delta_{BF} = -12.5^o$
δe = (Deg)
○ = -10
□ = - 7.5
◇ = 0
RH SSME
AFT STATION (X = 5")
PRESENCE OF NOZZLE
IN BASE AREA
PRODUCES HEAT
TRANSFER
AMPLIFICATION
OF ~ X10
h/h_{ref}
10^{-1}
10^{-2}
2×10^{-3}
0
25
50
75
100
125
Θ, Deg

Employing the heat transfer amplification magnitudes experienced by the SSME nozzles of the STS orbiter, estimates have been made of the local temperatures on the engine nozzle with and without body flap induced shock impingement. The heat transfer to the relatively quiescent flat base area has been estimated employing a flight test derived algorithm. The presence of the protruding nozzles results in an increase in heat transfer, as does the local impingement of body flap generated trailing shock systems. Local surface temperature predictions have been made based on a surface emittance of 0.8, a view factor to space of 0.5, and local radiation equilibrium. It is seen that trailing flap induced shock impingement on the nozzles clearly must be avoided.

A recommended approach would employ control flaps placed on the body rather than trailing. Technology development implications involve use of CFD, ground tests, and continuing evaluation of STS orbiter flight results.

SUMMARY OF ENGINE NOZZLE HEAT TRANSFER RESULTS

LOCATION	TEMPERATURE (^oF) FOR $\epsilon = 0.8, \mathcal{F}_A = 0.5$
RELATIVELY QUIESCENT BASE AREA	480
ON PROTRUDING ENGINE NOZZLE	1120
ON NOZZLE WITH SHOCK IMPINGEMENT	1770
ON NOZZLE WITH SHOCK IMPINGEMENT WITH X $2P_B$ UNCERTAINTY	2300

AOTV IMPLICATIONS

- o SHOCK IMPINGEMENT MUST BE AVOIDED
- o SUPPORTS CASE FOR MULTIPLE SMALL ENGINES
- o CONTROL FLAPS SHOULD BE ON BODY NOT TRAILING
- o TECHNOLOGY NEEDS:
  - APPLICATION OF CFD AND CALIBRATION OF METHODOLOGY
  - LOW REYs, HIGH M_∞ GROUND TESTS
  - CONTINUING EVALUATION OF STS RESULTS

3.3.2 Structural Analysis of Nozzles

A finite element model was developed for the analysis of the effects of a shuttle launch on an advanced LO_2/LH_2 engine nozzle structure. The finite element displacement model utilized Grumman's ASTRAL-COMAP (Automated Structural Analysis Comprehensive Matrix Algebra Program) in analyzing an advanced engine nozzle under the loading conditions imposed during an STS launch as defined by JSC 07700. The two nozzles studies were the Aerojet 3000 lb fixed nozzle engine, and the Rocketdyne 7500 lbf deployable nozzle engine. Only the Aerojet nozzle was modeled with finite elements.

The 1600 plate bending/membrane element model contains 1700 nodes with 6 degrees of freedom each. The fact that the nozzle is symmetric allowed the nozzle to be split for analysis purpose as shown on Page 221. In order to add an additional degree of safety to the analysis, the worst case boundary condition of a fixed cylinder end with a fixed nozzle was chosen. These conditions are represented pictorially in the Figure.

ANALYSIS OF ADVANCED LO_2/LH_2 ENGINE NOZZLES

- DISPLACEMENT BASED FINITE ELEMENT STRUCTURAL ANALYSIS USING ASTRAL-COMAP
  - AUTOMATED STRUCTURAL ANALYSIS COMPREHENSIVE MATRIX ALGEBRA PROGRAM
- EFFECTS OF ORBITER CARGO BAY ENVIRONMENT
  - ENVIRONMENT SPECIFIED IN JSC 07700
- AEROJET 3000 lb THRUST, $\epsilon = 1000:1$, FIXED NOZZLE
  - NOZZLE LENGTH = 66"
  - EXIT DIA = 30"
- ROCKETDYNE 7500 lb THRUST, $\epsilon = 970:1$, DEPLOYABLE NOZZLE
  - 2 SEGMENT NOZZLE
  - EXTENDED LENGTH = 112"
  - RETRACTED LENGTH = 60"
  - EXIT DIA = 52"

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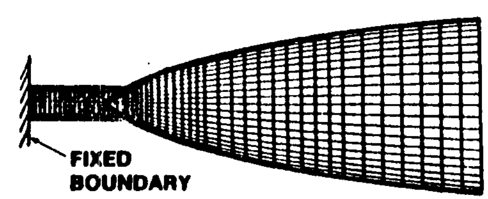
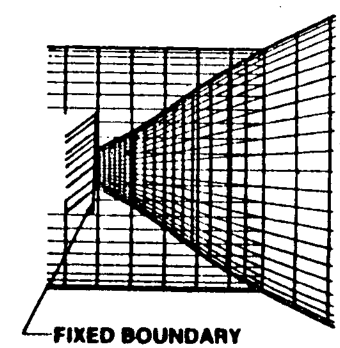
MODEL IDEALIZATION:

- 1600 PLATE BENDING/MEMBRANE ELEMENTS
- 1700 NODES; 6 DOF EACH
- SYMMETRIC/ANTISYMMETRIC LOADS AND BOUNDARY CONDITIONS EMPLOYED
- MODELS FIXED AT THROAT AND AT FORWARD END OF REDUNDANT CYLINDER
- RESULTS OF GENERIC MODEL ARE APPLICABLE TO BOTH NOZZLES

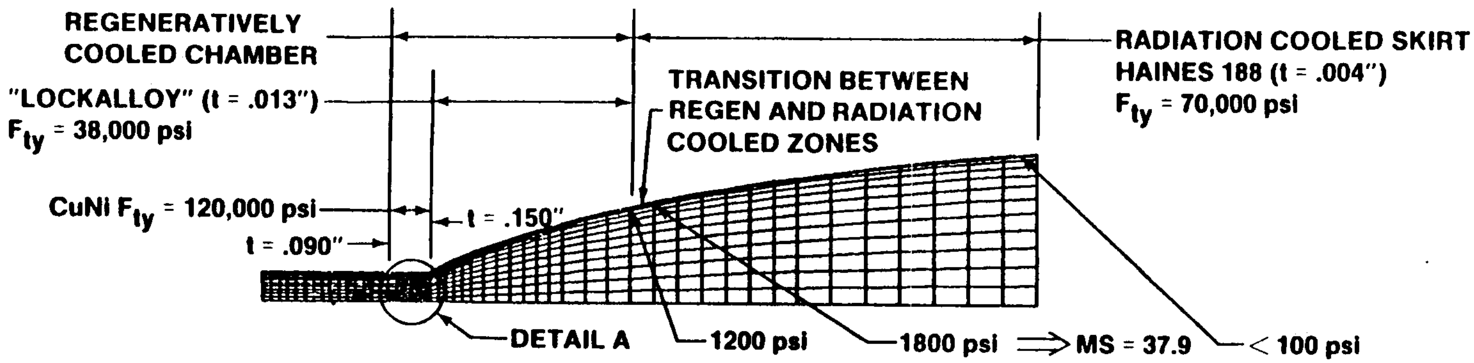


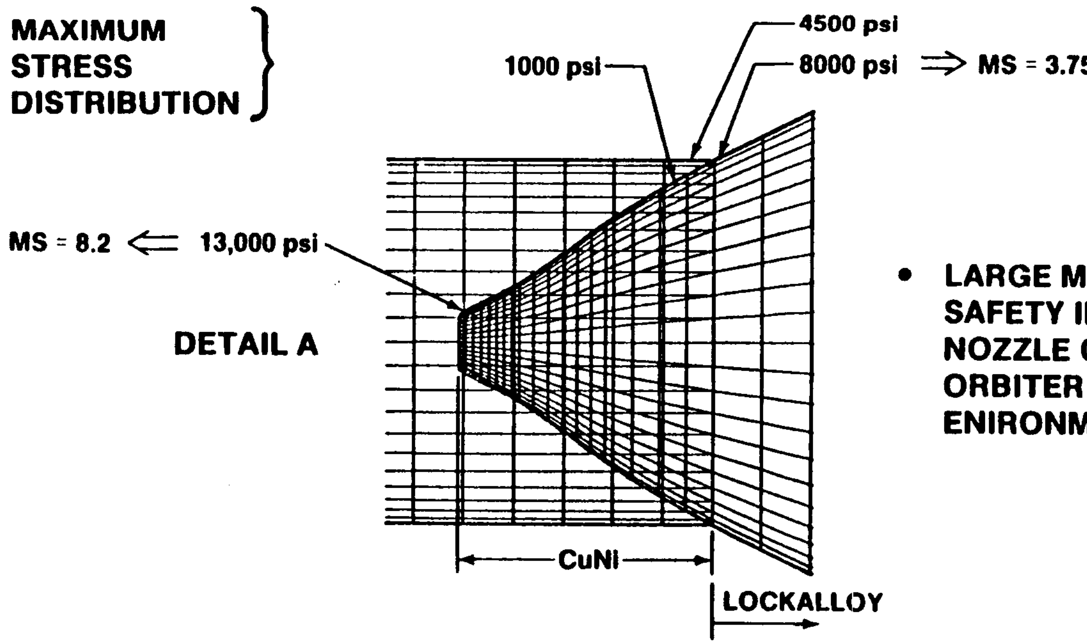


The results of the finite element analysis on the Aerojet 3000 lbf nozzle are shown in the Figure. All stresses are well below the allowable stresses of the nozzle materials, leading to high margins of safety. The first through fifth symmetric and antisymmetric vibration modes are substantially below the excitation environment induced by a shuttle launch. Therefore it is Grumman's opinion that the Aerojet nozzle is compatible with multiple shuttle launches without damage, based on the presently available nozzle design data.

RESULTS OF ANALYSIS OF AEROJET NOZZLE

- FUNDAMENTAL MODES { SYMMETRIC 3.59, 4.41, 4.67, 6.15 & 7.40 Hz; ANTI-SYMMETRIC 3.46, 4.15, 4.77, 6.07 & 7.61 Hz





- LARGE MARGINS OF SAFETY INDICATE NOZZLE OK FOR ORBITER CARGO BAY ENIRONMENT

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GRUMMAN

0283-011PP

The analysis of the retractable Rocketdyne 7.5K nozzle showed that the stresses in this larger nozzle are even smaller than in the Aerojet nozzle. The Rocketdyne nozzle should also be compatible with multiple reuse in a ground based/shuttle launched operation scenario.

RESULTS OF ANALYSIS OF ROCKETDYNE 7.5k LO_2/LH_2 NOZZLE

- ANALYSIS OF ROCKETDYNE NOZZLE DATA SHOWED LOWER STRESSES IN CRITICAL PORTION OF REGENERATIVELY COOLED NOZZLE SECTION THAN THE AEROJET NOZZLE
- NON-LOCALIZED STRESSES IN CARBON-CARBON PORTION OF ROCKETDYNE NOZZLE ARE WELL BELOW DESIGN STRESS

$$MS \geq \left[\frac{40{,}000}{200} - 1 \right] \geq 199$$

- NOZZLE IS SAFE IN ORBITER CARGO BAY ENVIRONMENT

GRUMMAN

0283-034PP

3.3.3 Engine Out Control Requirements

A series of interrelated propulsion issues were examined for advanced LOX/hydrogen engines. Specifically, we attempted to understand the effects of off-nominal factors on the kinds of biconic AOTVs we had found attractive. These topics are outlined in the figure.

Space vehicle center of gravity (cg) variations were examined to determine an expected or limiting case cg for a biconic AOTV. This cg location was used on conjunction with our vehicle configurations to determine engine gimbal angle requirements, or, for nongimbaled engines, performance degradation (loss of I_{sp}) from engine throttling.

PROPULSION ISSUES

- VEHICLE CG EXPECTATIONS
- H-IM PROPULSION CONSIDERATIONS
  - NOMINAL
  - ENGINE OUT
- GENERIC PROPULSION FOR MANRATED AOTV.

V84-1106-027(T)

A survey has been performed of the center of mass location tolerance of various spacecraft and entry vehicles designed and flown during the past two decades. These center of mass, CM, trends will be employed to determine the need/desirability for engine gimballing.

Historical data on the location of the cg of spacecraft from five corporations is shown in the figure:

o	Grumman	Lunar Module (Ascent Stage and Descent Stage)
o	General Electric	Re-entry Vehicles, Re-entry Vehicles Buses, and miscellaneous spacecraft ($\sim$ 2000 lb category)
o	TRW	Tracking and Data Relay Satellite, Fleet Satellite Communications Satellite
o	General Dynamics	Centaur G'
o	Rockwell	STS Orbiter

The strategic re-entry vehicles, RVs, surveyed ranged in mass form 70 to 350 lbs and CM location tolerances from 0.001 to 0.1 inches. These are "spinning" vehicles and require very accurate CM placement to provide impact accuracy by avoiding spinup or spindown during re-entry.

In the 1500 lb spacecraft category, both the spin stabilized Broadcast Satellite vehicles and the three axis stabilized Nimbus and Landsat series have a CM location tolerance 0.1 inches or less. The spin stabilized vehicles require low offset and cross inertia products to maintain stability. The attitude control fuel requirements is minimized for the three-axis controlled spacecraft by accurate CM locations.

The RV buses examined were in the 2K to 10K lbs range and worked with CM tolerances of up to $\pm$0.5 inches on the complete vehicle. These buses utilized gimballed engines for CM and inertia changes due to deployment of RV's when CM travel of an inch or two was allowed.

The Space Shuttle Orbiter CM tolerances are derived from NASA ICD 2-19001 for a 65K lb gross weight and are shown to verify a trend line for CM location tolerance vs. spacecraft weight.

The vertical axis of the Figure shows the cg displacement from a longitudinal axis of symmetry (X axis), thrusting line, or vehicle reference line. This displacement is measured in the radial direction such that

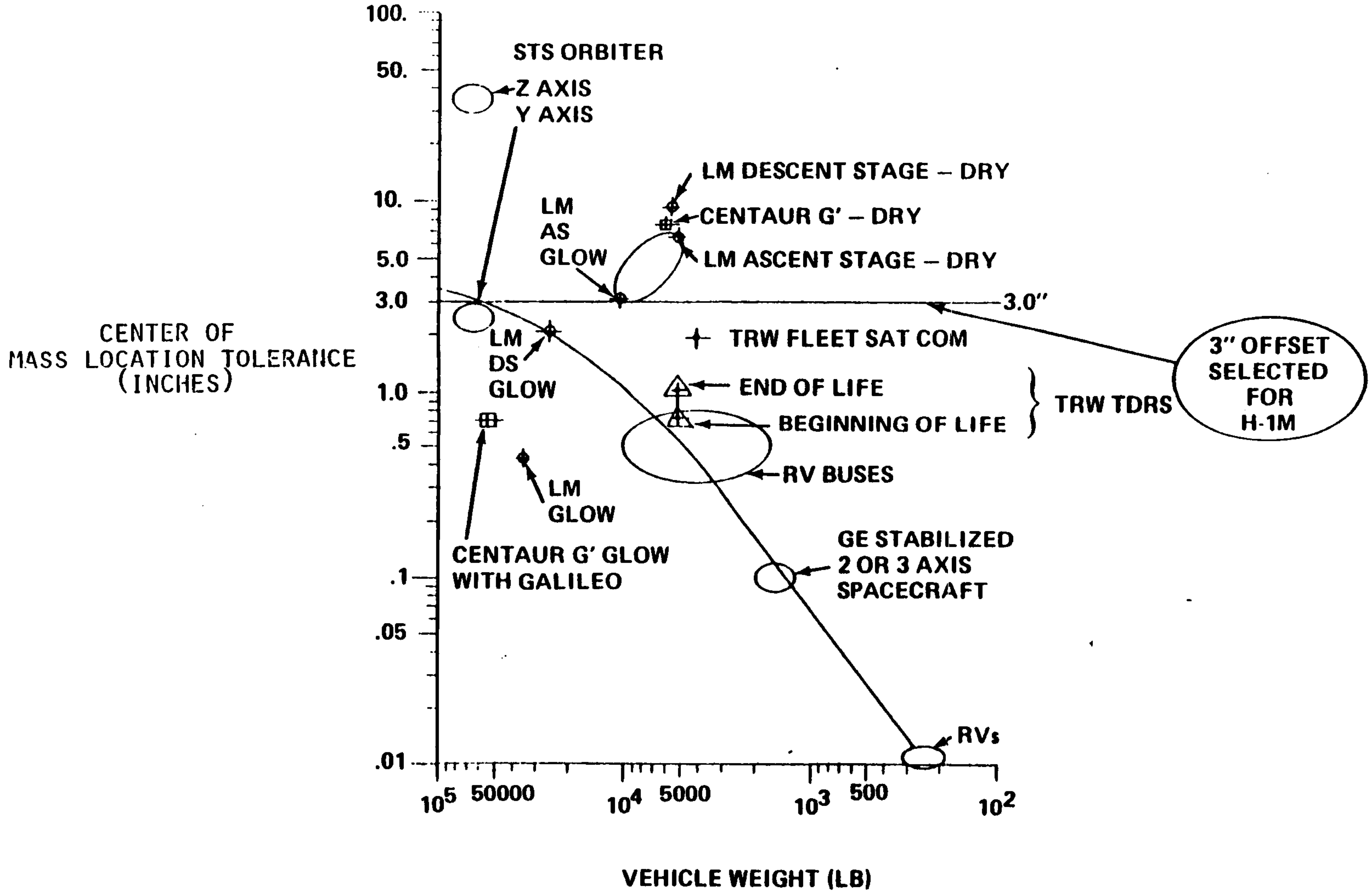
$$\delta_1 = \sqrt{Y^{2\prime} + Z^{2\prime}}$$

where Y' = cg displacement from X axis in Y direction
Z' = cg displacement from X axis in Z direction
δ_1 = radial displacement of cg from X axis

This historical data suggests that a 1" radial offset tolerance is a reasonable expectation for biconic AOTVs in the 5000 lb to 10,000 lb class. However, to fully explore the vehicle implication of cg displacements, we have selected the more conservative 3" displacement (which is highlighted in the figure) for our studies. This will allow some insight into how important (or unimportant) it is to maintain tight cg location control during the design phase.

CENTER OF MASS LOCATION TOLERANCE IN RADIAL DIRECTION

- HISTORICAL DATA FOR SOME SPACECRAFT & REENTRY VEHICLES



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V84-1106-032(T)

Redundancy considerations degrade vehicle performance through the addition of weight and/or the decrease in subsystem performance. In the H-1M vehicle, the reliability requirement of fail safe/fail safe necessitates the addition of components to the six fixed engine configuration. Operational conditions on the vehicle, such as a center of mass offset, also degrade vehicle performance. Throttling of the engines, which is required to correct the CM offset, cause a drop in the specific impulse of the engines and lower the initial T/W ratio of the vehicle. This induces greater gravity loss in the perigee burn. In the following pages, the effects of CM offset and propulsion system failure on the performance of the H-1M are investigated. The implications of corrective actions are compared in order to optimize the performance of the vehicle for the intended mission.

H-1M PROPULSION CONSIDERATIONS

PROBLEM

- CENTER OF MASS OFFSET AND REDUNDANCY CONSIDERATIONS AFFECT VEHICLE PERFORMANCE
  - CM OFFSETS CORRECTED BY THROTTLING DEGRADE SPECIFIC IMPULSE & THRUST/WEIGHT (HIGHER GRAVITY LOSS POSSIBLE)
  - SMALL BACKUP ENGINES NEEDED TO MEET REDUNDANCY REQUIREMENTS HAVE LOWER Isp THAN MAIN ENGINES

TOPICS:

- ESTABLISH CM OFFSET
- NOMINAL VEHICLE
- IMPLICATIONS OF ENGINE FAILURES

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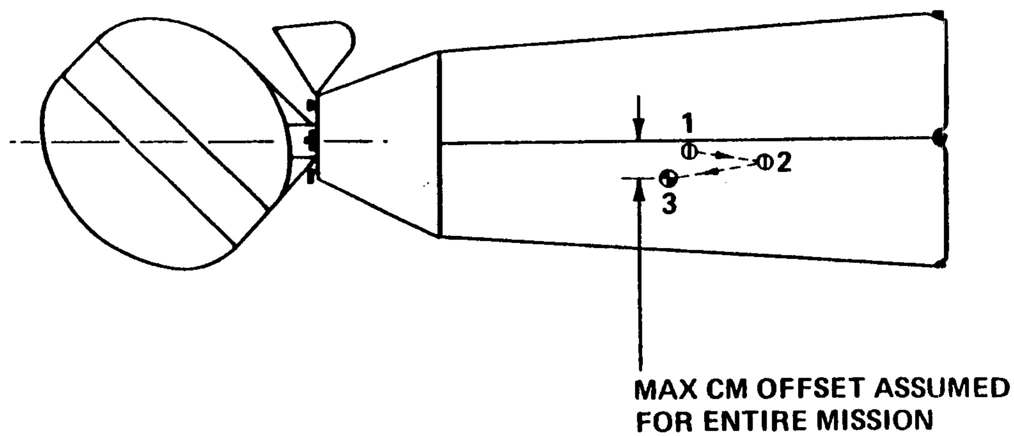
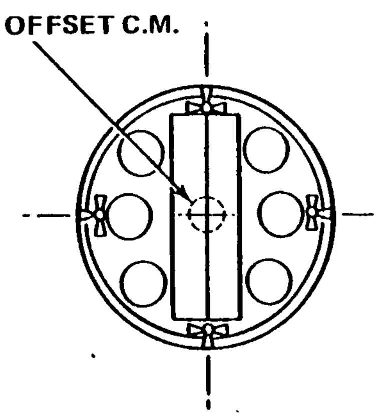
V84-1106-011(T)

The most efficient means of correcting for a radial CM offset under nominal operating conditions is throttling of selected engines. The weight penalties shown on the facing page represent the additional propellant required at start of mission to correct for the pitching moment which is produced by the CM offset. The diminished performance of the vehicle is caused by the decreased specific impulse on selected throttled engines, and, by the loss of total thrust caused by the throttling down of the engines. The increased gravity loss of the throttled vehicle requires the loading of additional propellant. The radial CM offsets stated on the facing page were assumed for the entire mission in the analysis, which produced the 60 lb and 120 lb penalties shown in the figure. Of the two, we believe the 60 lb penalty to be the more reasonable.

H-1M THROTTLING (NOMINAL)

- CENTER OF MASS OFFSET FROM VEHICLE CENTERLINE CORRECTED BY ENGINE THROTTLING
  - THROTTLING ENGINES CAUSES DEGRADATION IN ISP AND LOWER THRUST/WEIGHT
  - 60 LB EXTRA PROPELLANT REQUIRED FOR 3 IN. C.M. OFFSET AND 1 PERIGEE BURN
  - 120 LB EXTRA PROPELLANT REQUIRED FOR 6 IN. C.M. OFFSET AND 1 PERIGEE BURN
  - Tc = 3000 LBF, ϵ = 1000:1, 6 ENGINES
- SAME RESULTS OBTAINED WITH HINGED ENGINES





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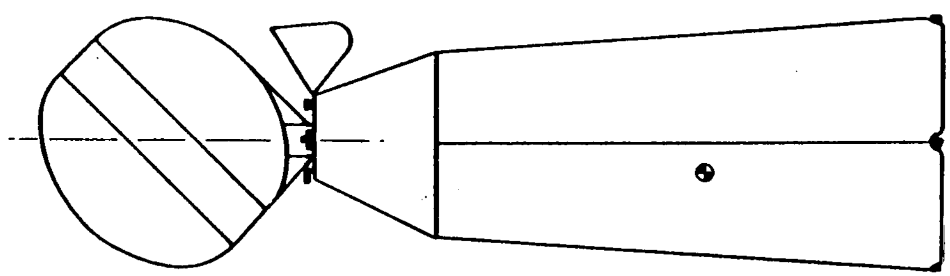
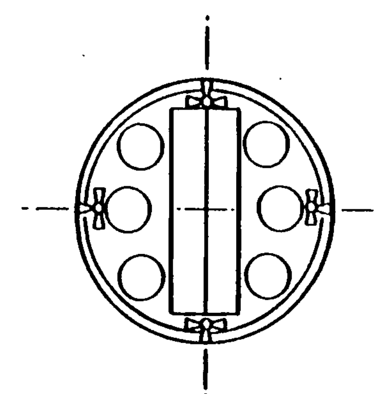
V84-1106-012(T)

The facing page lists the engine failure cases studied in the analysis. These conditions represent all the possible engine failure cases within the realm of the selected fail safe/fail safe design criteria.

IMPLICATIONS OF H-1M ENGINE FAILURES

- **ENGINE FAILURES STUDIED FOR H-1M**
  - — **ONE ENGINE OUT AT LEO**
  - — **ONE ENGINE OUT AT GEO**
  - — **TWO NON-ADJACENT ENGINES OUT AT GEO**
  - — **TWO ADJACENT ENGINES OUT AT GEO.**





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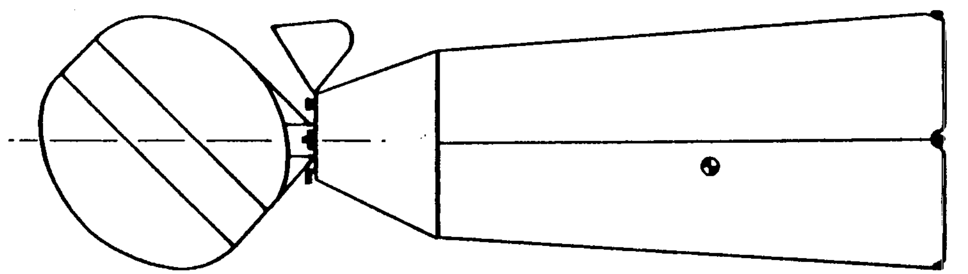
V84-1106-013(T)

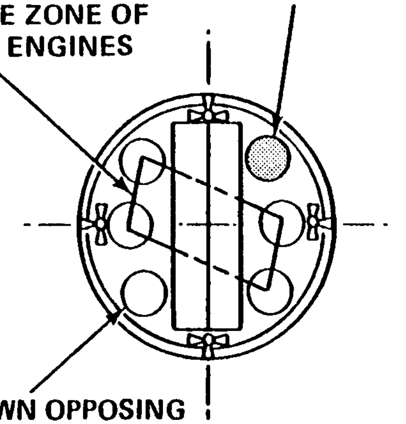
The first engine failure case studied is a single engine failure case. The failure type is handled by shuttling down the engine opposing the failed engine and throttling the remaining engines. The fail safe criteria can be met with the addition of 24 lbm of propellant at start of mission. The 24 lbm represents the additional amount of propellant required to deorbit the vehicle from GEO and return to LEO after a single engine failure.

A failop criteria can be met with 300 lbm of additional propellant and 20 lbm of engine hinge hardware. This would allow the vehicle to perform the Grumman recommended manned design reference mission (8K up, 6K back) after a single engine failure at any point during the mission. A decision has not been made as to whether a failop/fail safe rating on the engines is worth the additional weight over the fail safe/fail safe engine rating.

ONE ENGINE OUT OPERATION OF H-1M

- SHUT DOWN OPPOSING ENGINE AND ALIGN THRUST VECTOR BY THROTTLING REMAINING ENGINES
- PROPELLANT PENALTY DURING ENGINE OUT OPERATION WITH 3 IN C.M. OFFSET
  - 24 LB FROM GEO
  - 424 LB FROM LEO WITH SINGLE PERIGEE BURN AND THROTTLING FIXED ENGINE
  - 300 LB FROM LEO WITH SINGLE PERIGEE BURN AND HINGED ENGINES (+20 LB FOR HINGE HARDWARE)





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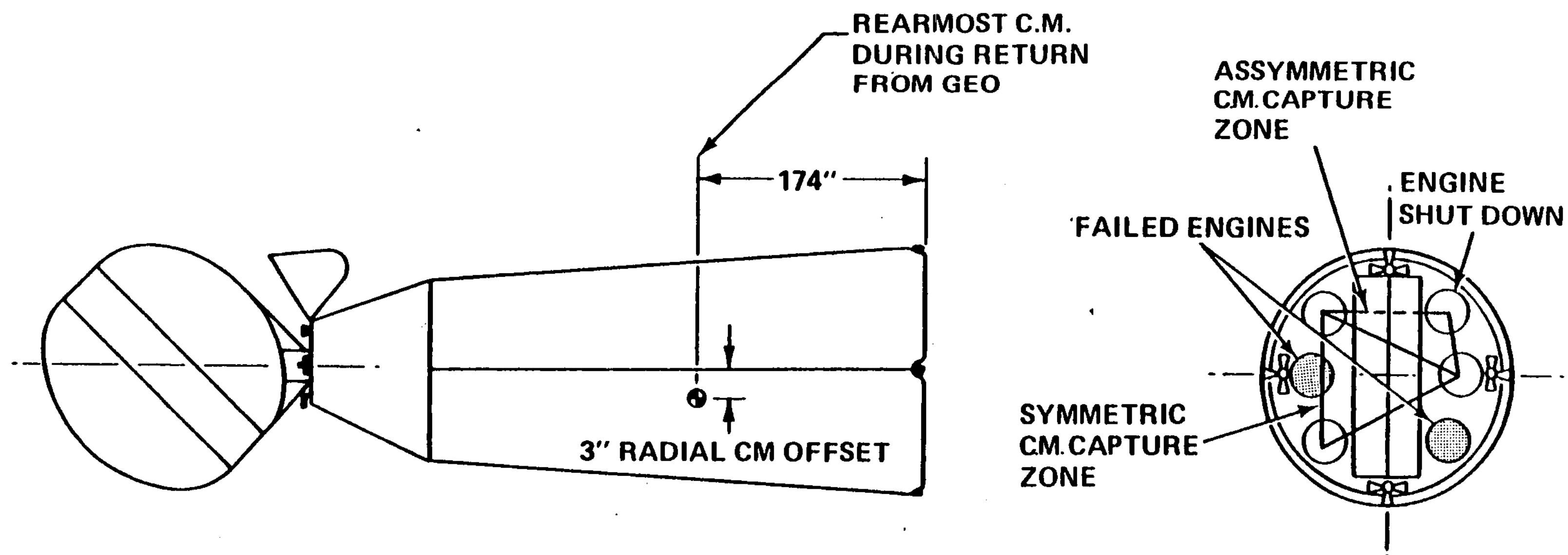
V84-1106-014(T)

The problem of two non-adjacent engine failure at GEO can be corrected by shutting down of a selected engine and throttling the remaining engines. As long as the AOTV center of mass remains inside the polyhedron formed by connecting the centerlines of operational engines (a triangle in the figure), the vehicle is stable during main propulsion system burns if the engines are continuously throttleable. The extra propellant required for compensation of the decreased efficiency of the throttled engines is 64 lbm of additional propellant at start of mission for an assumed 3" max radial offset.

TWO ENGINE OUT OPERATION OF H-1M

NON-ADJACENT ENGINES

— TWO NON-ADJACENT ENGINE FAILURES CAN BE CORRECTED BY THROTTLING THREE FIXED ENGINES
  - EXTRA PROPELLANT REQUIRED FOR 3 ENGINE RETURN FROM GEO WITH 3 IN C.M. OFFSET = 64 LB
  - NO GIMBALING OF ENGINES REQUIRED FOR NON ADJACENT TWO ENGINE OUT RETURN FROM GEO
  - NO ADDITIONAL ACS PROPELLANT REQUIRED FOR RETURN FROM GEO.



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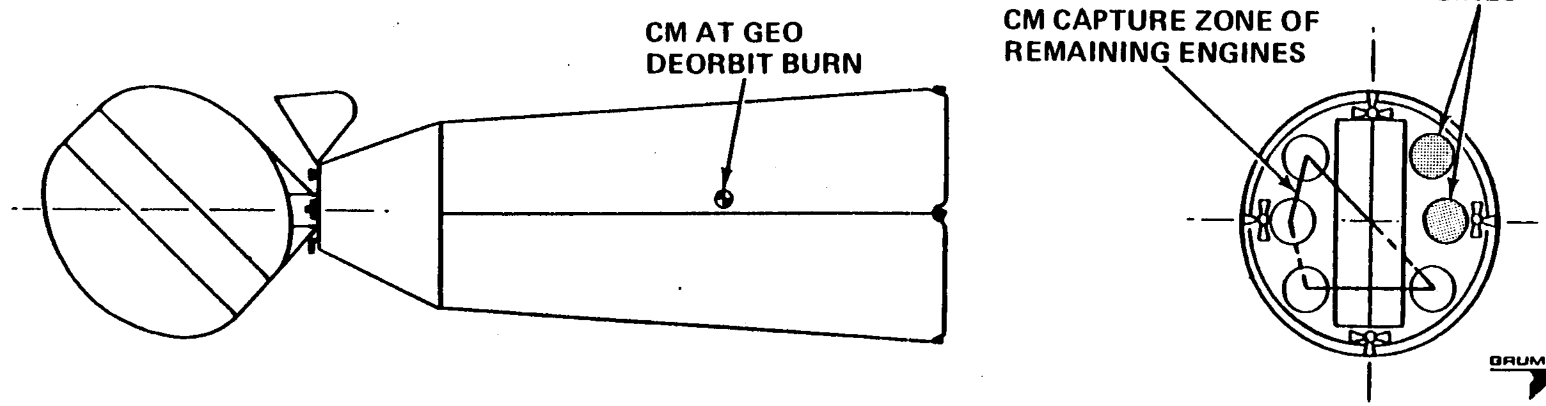
V84-1106-015(T)

The worst case scenario for the fixed engine H-1M is the failure of two adjacent engines with the CM outside of the capture zone of the remaining engines. In the event that this low probability occurrence is a reality, the CM of the vehicle cannot be brought in line with the resultant thrust vector by fixed main engines alone. There are a number of possible corrective actions for this condition and these corrective actions will be evaluated in the following pages. All corrective actions require continuous throttling of the main engines to track the movement of the center of mass.

TWO ENGINE OUT OPERATION OF H-1M (CONT)

ADJACENT ENGINES

- WORST CASE IS LOSS OF ADJACENT ENGINES WITH C.M. NOT IN CAPTURE ZONE OF REMAINING ENGINES
- PROBABILITY OF OCCURANCE = .000003866, 1/5 THE PROBABILITY THAT ANY 2 ENGINES FAIL
- MOVEABLE MASS, ACS OR HINGED ENGINES ARE REQUIRED TO ALIGN THRUST VECTOR WITH OFFSET C.M. IN WORST CASE CONDITION
- CONTINUOUS THROTTLING OF ENGINES REQUIRED TO TRACK C.M.



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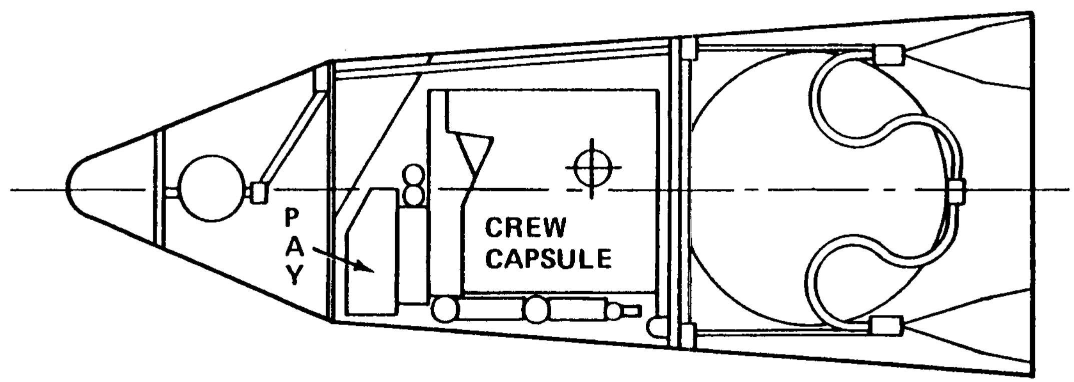
V84-1106-016(T)

A possible solution to the two adjacent engine out worst case CM location problem is the movement of a large mass inside the vehicle. This translation of the mass would have to be of large enough magnitude to move the CM into the capture zone of the remaining engines. In the case of the H-1M vehicle, as shown on the facing page, the translation of a large mass (crew capsule) inside the aeroshell is not sufficient to correct a worst case very large CM offset without growth of the aeroshell. Growth of the aeroshell beyond its present proportions would add vehicle weight. Other solutions to the CM thrust vector alignment problem are investigated in the following pages.

BALANCE THRUST/CM OFFSET WITH MOVABLE MASS?



- POSSIBLE BACK UP SYSTEM FOR 2 ENGINES OUT AT GEO & CM IN UNFAVORABLE LOCATION
- MOVE LARGE INTERAL MASS TO RETURN CM OFFSET TO VEHICLE CENTERLINE
  - H-1M GEO DEORBIT WT ≈ 21000 LB
  - MANNED CAPSULE WT ≈ 5300 LB

	3" CM OFFSET	1" CM OFFSET
REQ'D CAPSULE RADIAL MOTION	$\sim \frac{21000\,(3'')}{5300} = 12''$	$\sim \frac{21000\,(1'')}{5300} = 4''$
INCREASED AEROSHELL DIA	$\sim \frac{[140'' + 2\,(12'')]\,\pi}{140\pi} \Longrightarrow + 17\%$	$\sim \frac{[140'' + 2\,(4'')]\,\pi}{140\pi} \Longrightarrow + 6\%$

RESOLUTION AWAITS TRADEOFF BETWEEN FLAPS, ACS & MOVING MASS

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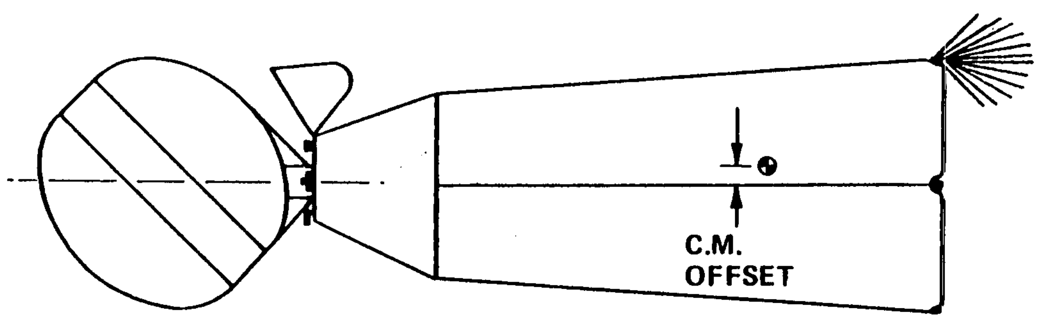
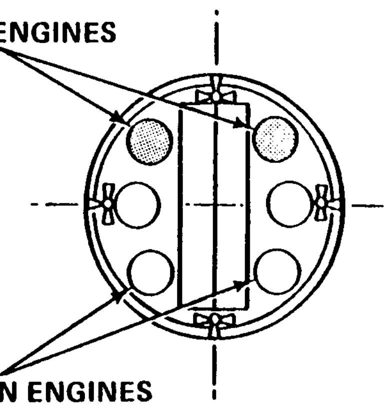
V84-1106 U28(T)

A method for compensating for CG/thrust vector misalignment is the use of rear facing oversize ACS thrusters. This requires that extra ACS propellant be carried on all missions in the event of a two adjacent engines failure scenario. This extra ACS propellant capacity requires additional cryogenic propellant because of the weight increase of the vehicle. The total penalty for this method of thrust vector alignment is 785 lbm. This figure includes the weight penalties for the oversized thrusters, additional plumbing weight and the additional cryogenic and bipropellant mass required for fail safe/fail safe return from GEO.

AXIAL ACS vs THRUST PITCHING MOMENT

- THRUST VECTOR/C.M. ALIGNMENT BY OVERSIZED BYPROP HYD REAR-FACING THRUSTERS
  - ACS PROPELLANT ADDS TO TOTAL IMPULSE
- N_2O_4/MMH
  - ISP = 300 LBF SEC/LBM IN STEADY STATE MODE
  - T $\approx$ 100 LBF MAX
  - 740 LBM OF ADDITIONAL PROPELLANT REQUIRED (CRYO + BYPROP HYD) FOR RETURN FROM GEO WITH TWO ADJACENT ENGINE FAILURES AND ADVERSE C.M. CONDITION
  - 45 LBM FOR OVERSIZED THRUSTERS AND ADDITIONAL PLUMBING
  - TOTAL PENALTY = 785 LB.





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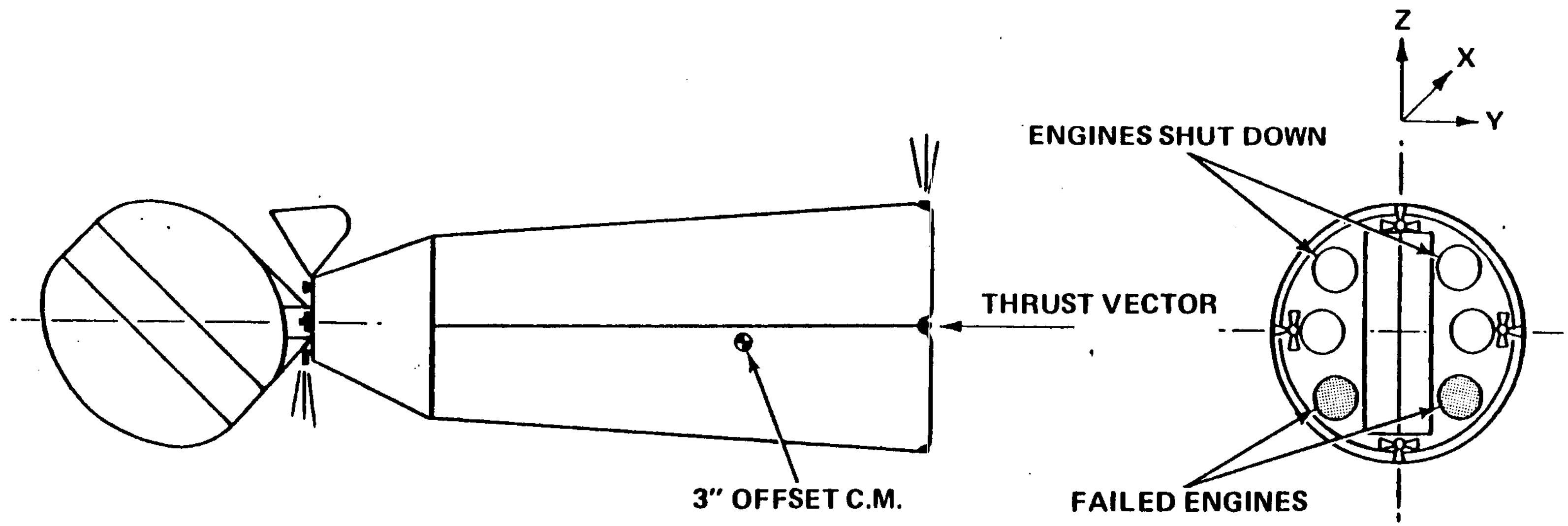
V84-1106-017( )

A third method of correcting for thrust vector/CM misalignment under two adjacent engine out conditions in the H-1M vehicle is the use of Y-Z plane thrusters during the main propulsion burns. The Y and Z plane thrusters are used for moment compensation. The weight penalty at liftoff for availability of this operational mode is 590 lbm. This represents the weight of the additonal bipropellant required for a two engine out corrective burn as well as the additional cryogenic propellant required for transport of the additional tankage and propellant mass in the ACS system.

RADIAL ACS vs THRUST PITCHING MOMENT

- 590 LBM OF ADDITIONAL PROPELLANTS (CRYO + BYPROP HYD) REQUIRED FOR RETURN FROM GEO FOR TWO ADJACENT ENGINE OUT CONDITION WITH 3 IN. C.M. OFFSET
  - ISP BYPRO HYD = 300 LBF SEC/LBM IN STEADY STATE MODE
- THRUST ALIGNMENT BY ACS Y/Z PLANE THRUSTERS REQUIRES LARGE PROPELLANT QUANTITIES



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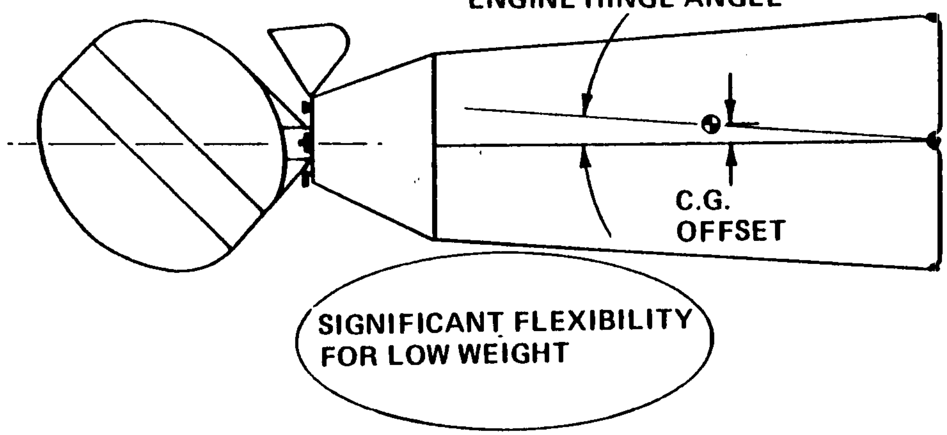
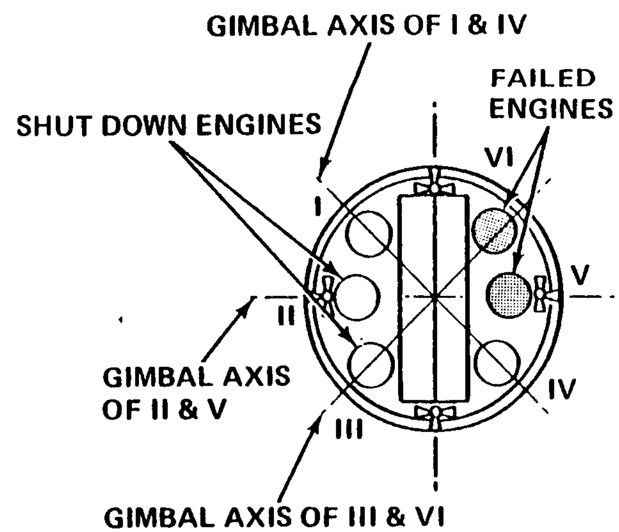
V84-1106-018(T)

The final solution investigated for handling the worst case non-catastrophic failure of two engines is the hinged engine concept. The engines rotate on a axis which passes through both the engines centerline and the vehicle centerline. As shown on the facing page, the failure of two adjacent engines is compensated for by shutting down the engines opposite the failed engines and rotating the two firing engines on their hinge axes until the resolved thrust vector of the engines passes through the CM of the vehicle. Twenty pounds of additional hardware are required to change the fixed engines to hinged engines (with a total motion of 4^{o}). Specific impulse degradation under throttling cause 70 lbm of additional main propulsion system propellant consumption. The total weight penalty for this concept at start of mission is 90 lbm.

GIMBAL ENGINES ON ONE AXIS: HINGED ENGINES

- ENGINE ROTATION IS PERPENDICULAR TO RADIAL LINE (PARALLEL TO SHELL)
- HINGING + THROTTLING PRODUCES FAILSAFE/FAILSAFE AOTV FOR ALL C.M. LOCATIONS
- 20 LB HARDWARE REQUIRED FOR HINGING $\pm 2^{\circ}$
- 70 LB FOR EXTRA PROPELLANT AND DEGRADED ISP DUE TO THROTTLING
- TOTAL PENALTY = 90 LB





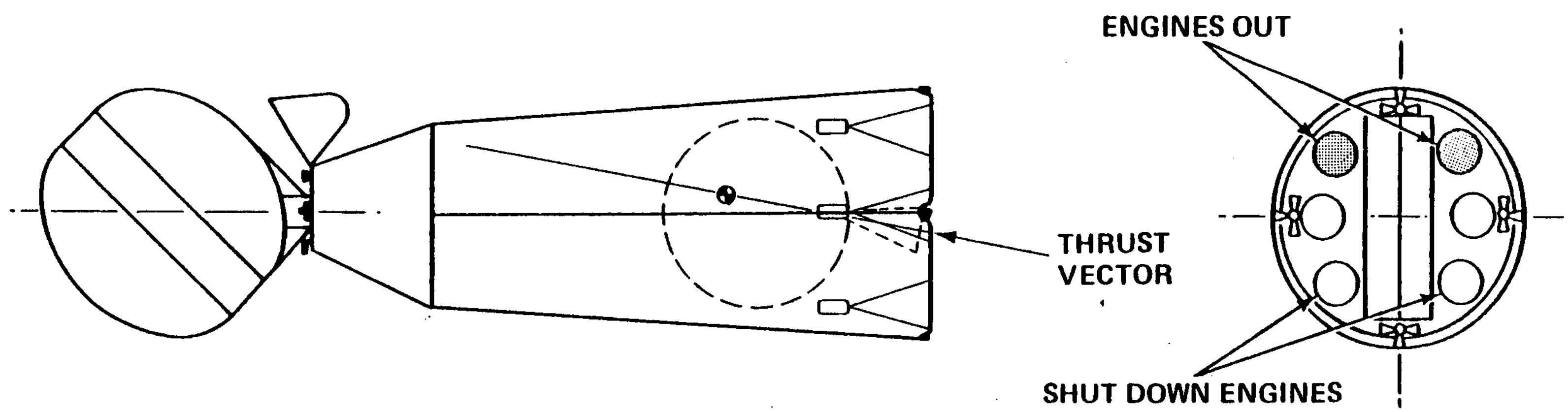
GRUMMAN

V84-1106-019(T)

The hinged engine concept on the H-1M vehicle was found to be the most attractive method of providing fail safe/fail safe operation of the engine subsystem. Because the hinge angles required for thrust vector alignment under all two engine out conditions are small, the engine location remains unchanged from the six fixed engine H-1M and maintains the packaging advantages for biconic AOTVs of this propulsion system approach.

CONCLUSIONS OF H-1M ENGINE FAILURE IMPLICATIONS

- HINGED ENGINES ARE THE PREFERED SOLUTION FOR A FAILSAFE/FAILSAFE SIX ENGINE CONFIGURATION
  - SMALL HINGE ANGLE MAINTAINS THE PACKAGING ADVANTAGES OF THE SIX ENGINE CONFIGURATION
  - WEIGHT PENALTY FOR MEETING THE FAILSAFE/FAILSAFE CRITERIA IS SMALL IN COMPARISON TO ALL OTHER METHODS



GRUMMAN

V84-1106-020(T)

For an extreme maximum radial center of mass offset of 3", the combination of hinging and throttling allow the compensation for all engine out cases if a sufficient amount of propellant is available. With the hinged engine concept, fail safe/fail safe performance can be had for 20 lb in additional hardware weight combined with an additional propellant reserve of 70 lb. Failop/fail safe performance requires the addition of 230 lb of propellant over and above the required propellant mass for the fail safe/fail safe system. This addition of propellant would allow for the completion of a mission after a single engine failure at any time in the mission.

The addition of the hardware for hinging gimbals and actuators, and the required clearances, require no growth and only minor packaging changes in the H-1M.

SUMMARY OF H-IM PROPULSION CONSIDERATIONS

- NOMINAL ENGINE OPERATIONS
  - 3" RADIAL CM OFFSET REQUIRES 60 LB OF ADDITIONAL PROPELLANT FOR THROTTLING LOSSES
- ENGINE OUT OPERATIONS
  - 1 ENGINE OUT AT GEO → 20 LB OF EXTRA PROPELLANT
  - 2 ENGINES OUT AT GEO → { 70 LB OF EXTRA PROPELLANT; 20 LB OF HINGING HARDWARE }
  - 1 ENGINE OUT AT LEO
    - SINGLE PERIGEE BURN AND THROTTLING FIXED ENGINES REQUIRES 420 LB OF ADDITIONAL PROPELLANT
    - SINGLE PERIGEE BURN AND HINGED ENGINES REQUIRES 300 LB OF ADDITIONAL PROPELLANT.

GRUMMAN

V84-1106-029(T)

3.3.4 Number of Engines and Thrust Level(s)

A generic propulsion system trade was performed to determine the influence of the number of main engines while adhering to fail safe/fail safe reliability design goal (on vehicle configurations with one through six engines). The baseline vehicle for the study was a 14.5′ diameter manned biconic OTV with an aft spherical LO_2 tank, external LH_2 tank, current state of the art nozzle protection by the vehicle aeroshell, and rear facing main engines. all engines had an expansion ratio of 1000/1 and total thrust (under nominal conditions) of 15,000 lbf for each vehicle.

All configurations were designed to a fail safe/fail safe level on the main propulsion system. For the one and two engine configurations, small (500 lbf) reserve engine(s) were added along with additional plumbing to bring the engine subsystem to a fail safe/fail safe level. The subsystem trade was run for 1 through 6 main engines. It assumed that two independent propulsion system failures took place at GEO. These non-catastrophic worst case failures required the vehicle to return the crew to LEO on a backup system or by the use of redundant components.

GENERIC PROPULSION TRADE FOR FAILSAFE/FAILSAFE MANRATED AOTV

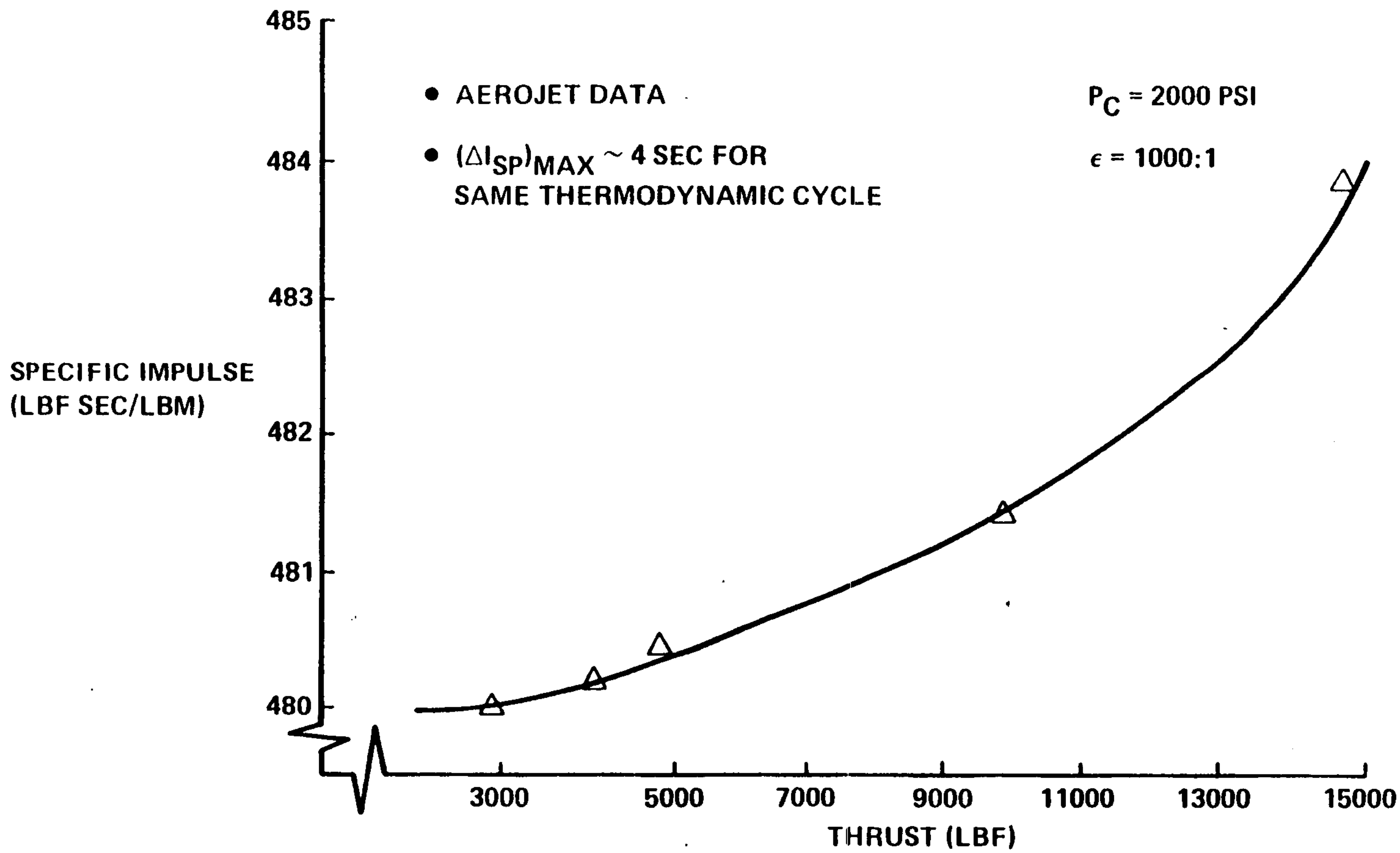
- FOR ALL VEHICLES: $T_{TOTAL} = 15000$ LBF, & $\epsilon_{MAIN} = 1000:1$
- VARY MAIN ENGINE NUMBER FROM ONE TO SIX
- USE OF T = 500 LBF, ISP = 476.7 SEC, REDUNDANT ENGINE(S) WHERE APPLICABLE
- ALL VEHICLES MUST BE ABLE TO SUSTAIN TWO INDEPENDENT FAILURES (INCLUDING ENGINE COMPONENTS) AND RETURN CREW FROM GEO TO LEO
  - ANALYSIS ASSUMES BOTH FAILURES TAKE PLACE AT GEO. CREW IS RETURNED ON BACKUP SYSTEM
- BASELINE VEHICLE USED FOR ENGINE GIMBAL ANGLES WAS 40' LONG BICONIC WITH EXTERNAL HYDROGEN TANK.

GRUMMAN

V84-1106-021(T)

The facing page illustrates the engine trend data used in this study. The data supplied by Aerojet Techsystems Comany illustrates (for the same thermodynamic cycle) chamber pressure, expansion ratio, and the functional relationship between single engine thrust and specific impulse for an advanced LO_2/LH_2 engine. For the range of 2500 lbf to 15,000 lbf, the change in specific impulse is approximately 4 lbf sec/lbm. Data provided by Rocketdyne also defined a specific impulse of 476.7 for a 500 lbf advanced LO_2/LH_2 engine.

SPECIFIC IMPULSE vs THRUST LEVEL FOR AN ADVANCED CRYO OTV ENGINE



GRUMMAN

V84-1106-024(T)

The table on the facing page presents calculated values for the parameters considered in this tradeoff study. All configurations, with the exception of the baseline 6 fixed engines, meet the fail safe/fail safe criteria.

The first row represents a specified propellant residual and reserve allowance. The second row defines the maximum propellant that become unobtainable by the propellant acquisition device under gimbal requirements caused by the worst case engine out condition. Redundant engine and line hardware (third row) are required in the one and two engine configurations to bring the vehicle to fail safe/fail safe levels in the engine subsystem. The fourth row, Δweight Phase I, represents the weight differences in the propulsion/pressurization systems from a 6 engine baseline and is taken from the Phase I AOTV report of this study contract. The "Gimbals and Actuators" row is an estimate of the weight of the gimbal drives required for thrust vector/CM alignment for each of the tradeoff configurations. The next row "Redundancy Simplification from Phase I", illustrates the weight savings from the Phase I values in the areas of valve and line redundancy. These were made possible by the addition of backup engine(s) in the one and two engine configurations. The bottom two rows are comparisons between the baseline (6 fixed engines) and alternative concepts.

RESIDUAL PROPELLANT & PROPULSION SYSTEM TRADE

- FAILSAFE / FAILSAFE

	NUMBER OF ENGINES	1	2	3	4	5	6 (HINGED)	6(FIXED)*
PROPELLANT	RESIDUALS & RESERVES (lb)	1500	1500	1500	1500	1500	1500	1500
PROPELLANT	OFF AXIS THRUSTING RESIDUALS (lb)	50	95	95	25	20	0	N.A.
PROPULSION HARDWARE	REDUNDANT ENGINE AND LINE HARDWARE (lb)	350	145	0	0	0	0	0
PROPULSION HARDWARE	Δ WEIGHT PHASE I (lb)	- 246	- 243	- 260	-173	- 86	0	0
PROPULSION HARDWARE	GIMBALS & ACTUATORS (lb) / TOTAL GIM. ANG.	40 / 5°	180 / 22°	170 / 20°	140 / 16°	128 / 14°	20 / 4°	0 / 0
PROPULSION HARDWARE	REDUNDANCY SIMPLIFICATION FROM PHASE I (lb)	- 102	- 87	0	0	0	0	0
PROPULSION HARDWARE	Σ OF PROPELLANT + PROPULSION (lb)	1592	1590	1505	1492	1562	1520	1500
	Δ WEIGHT FROM 6 FIXED ENGINES (lb)	92	90	5	- 8	62	20	0

*CONFIGURATION DOES NOT MEET FAILSAFE / FAILSAFE CRITERIA FOR ALL CM POSSIBILITIES

GRUMMAN

V84-1106-022(T)

The facing page defines the pertinent subsystem weights (in pounds) of the three cases considered in this study: A, B and C. In the upper table, Case A uses the values for change in AOTV aeroshell weight which were obtained in Phase I of this study. Case B defines an AOTV with an exposed nozzle during flight. It assumes the AOTV aeroshell weight is independent of the nozzle length change associated with varying engine configurations. Case C is the true functional relation between engine nozzle length and weight and is unknown at this time (TBD - to be determined). Case C lies somewhere between cases A and B. An estimate of Case C is shown on Pages and .

Using these aeroshell weight values, the valves for the propulsion subsystem taken from the previous figure (Row II in upper table, and a baseline weight for the reference configuration (Row I in upper table), the values in the lower table were generated. The lower table is a tabular summation of the burnout weight, and delta burnout weight from the reference configuration, for the tradeoff configurations. The values are used in the scenario illustrated in the next figure (Page) to generate GLOWS for the tradeoff configurations (Page).

ENGINE NUMBER WEIGHT TRADE RESULTS

	NUMBER OF ENGINES	1	2	3	4	5	6 HINGED	6 FIXED
I	BASELINE AOTV (LB)	14200	14200	14200	14200	14200	14200	14200
II	Σ OF PROPELLANT & PROPULSION SYSTEM (LB)	1592	1590	1505	1492	1562	1520	1500
A	Δ AEROSHELL WEIGHT OF PHASE I (LB)	393	373	213	140	6	0	0
B	LENGTH INSENSITIVE VEHICLE Δ WT (LB)	0	0	0	0	0	0	0
C	Δ IN AEROSHELL WEIGHT-REFINED	← TBD →						

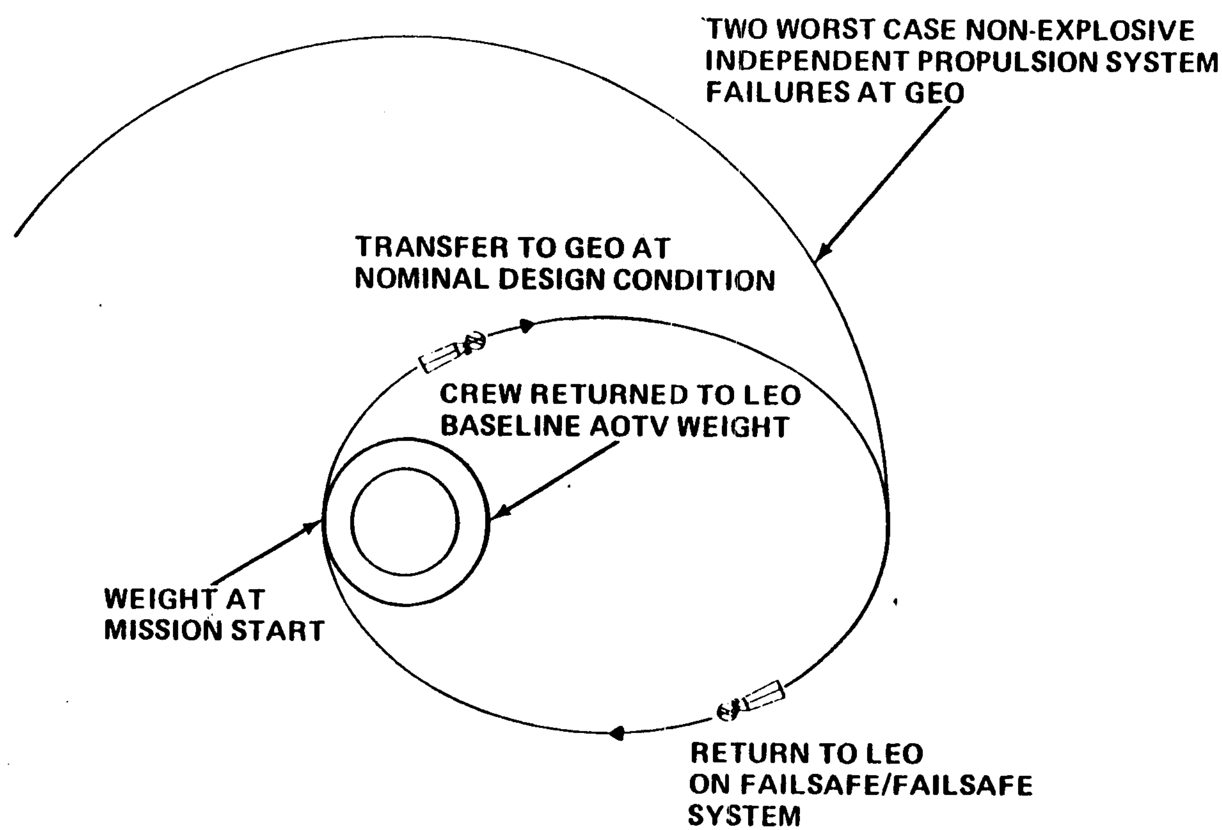
NUMBER OF ENGINES		1	2	3	4	5	6 HINGED	6-FIXED
AEROSHELL CASE A I + II + A (LB)	W	16185	16163	15918	15832	15768	15720	15700
	ΔW	485	463	218	132	68	20	0
AEROSHELL CASE B I + II + B (LB)	W	15792	15790	15705	15692	15762	15720	15700
	ΔW	92	90	5	−8	62	20	0
AEROSHELL CASE C I + II + C (LB)	W / ΔW	← TBD →						

GRUMMAN

V84-1106-023(I)

The selected failure scenario for our engine number trade study assumes two worst case non-catastrophic engine subsystem failures at GEO. Return from GEO to LEO is accomplished by the use of backup propulsion components or redundant components (in the case of multiple engine configurations).

PROPULSION SYSTEM TRADE
FAILURE SCENARIO SELECTED FOR ANALYSIS



GRUMMAN

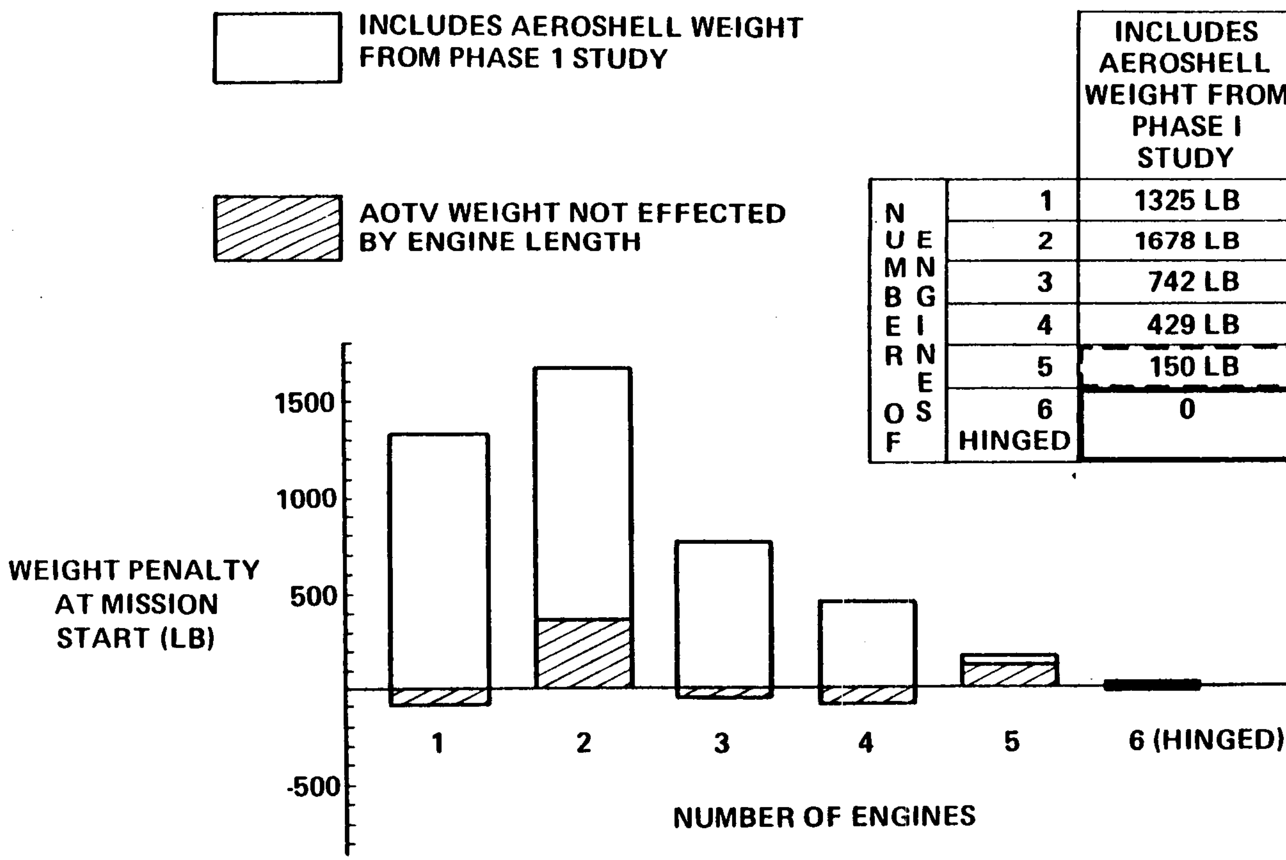
V84-1106-025(T)

Shown in the adjacent figure are the results of our generic propulsion system trade study. The first column of the table shows the difference from the new reference concept (6 hinged engines) in gross liftoff weight for all tradeoff configurations based on the aeroshell weight studies conducted in Phase I of this AOTV System Technology Analysis Study. The lowest weight configuration that meets the set reliability requirement of fail safe/fail safe for state of the art thermal protection and engine nozzle protection is the six hinged engine configuration. For the case of aeroshell weight independent of engine length, the one engine configuration and the 4 engine configuration are the combinations with the lowest glow, but there are two other equally attractive configurations (6 hinged and 3 engines) within the uncertainty range of the study. The results of this tradeoff are also shown in graphical form in the figure.

WEIGHT PENALTY AT START OF MISSION vs NUMBER OF ENGINES

- FAILSAFE/FAILSAFE AOTV
- 1500 LBF TOTAL THRUST
- AEROJET ENGINE DATA
- 3" C.M. OFFSET EFFECTS
  - ENGINE THROTTLING PENALTIES + INCREASED GRAVITY LOSS INCLUDED
  - SINGLE PERIGEE BURN, SINGLE STAGE TO GEO & BACK.

NUMBER OF ENGINES	INCLUDES AEROSHELL WEIGHT FROM PHASE I STUDY	AOTV WEIGHT NOT EFFECTED BY ENGINE LENGTH
1	1325 LB	-70 LB
2	1678 LB	346 LB
3	742 LB	-17 LB
4	429 LB	-70 LB
5	150 LB	128 LB
6 HINGED	0	0



GRUMMAN

V84-1106-030(T)

The major conclusions we derived from our generic propulsion system trade are listed on the facing page.

SUMMARY OF GENERIC PROPULSION TRADE

- FOR FAIL SAFE/FAIL SAFE AOTV
  - 15000 LB TOTAL THRUST
  - $\epsilon = 1000:1$
  - 480 SEC $\leqslant I_{SP} \leqslant$ 484 SEC

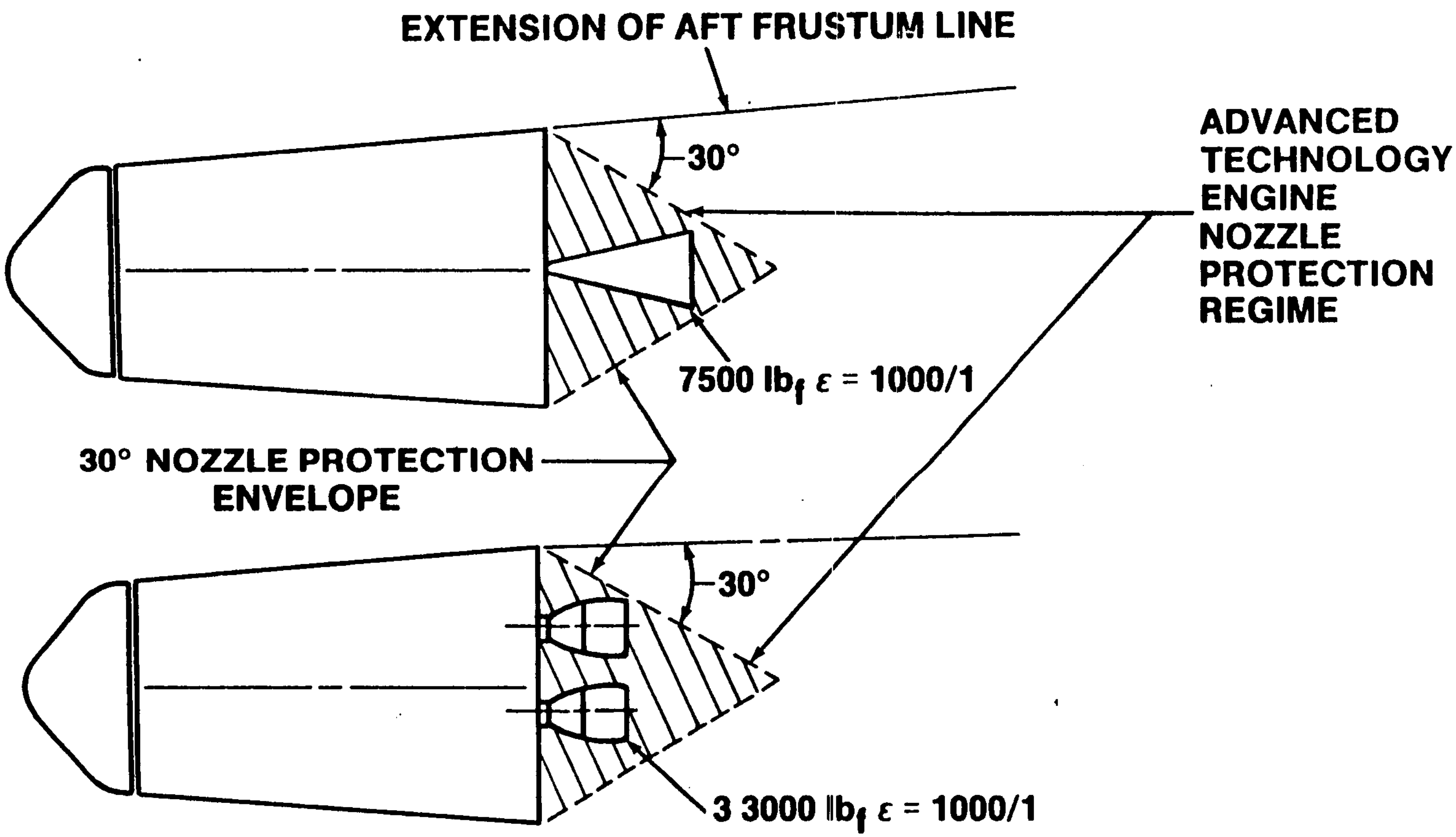
  ADVANCED TECHNOLOGY LOX/H ENGINES
- ENGINE GIMBAL ANGLES: 5° TO 22°; HINGED ANGLES: 4°
- FOR AOTV WEIGHT NOT EFFECTED BY ENGINE LENGTH
  - 1, 3, 4 & 6 ENGINES PRODUCE APPROXIMATELY EQUAL AOTV WTS. AT THE START OF THE MISSION
- FOR AOTV WEIGHT OF PHASE I STUDY (80 LB/FT)
  - 6 HINGED OR FIXED ENGINES PRODUCE A SIGNIFICANTLY LIGHTER AOTV AT THE START OF THE MISSION THAN 1, 2, 3 OR 4 ENGS
- FOR BICONIC VEHICLES, ACCURATE TRADE STUDY RESULTS AWAIT FURTHER DEFINITION
  - HYPERSONIC HEATING PROTECTION
  - BETTER DEFINITION OF TPS/STRUCTURE WEIGHTS.

GRUMMAN

V84-1106-031(T)

An advanced engine nozzle protection regime for space based vehicles is illustrated. The safe region is defined by a cone generated by a 30° angle inward off an extention of the rear frustum of the vehicle. The vehicles shown in the figure illustrate that both large single engines and smaller engine clusters can be entirely within the "safe cone" envelope. Flight test verification of the size of a "safe cone" envelope is required for implementation into a reusable AOTV program.

BICONIC AFT ENGINE DESIGN OPTION



- VEHICLE WEIGHT SAVINGS WITH CENTRALLY LOCATED ENGINES
- FLIGHT TEST DATA <u>REQUIRED</u> BEFORE THIS DESIGN TECHNIQUE CAN BE USED

GRUMMAN

0283-012PP

The figure repeats the tables of Page . The new elements in this figure are the TBD values of the figure on Page , which have been replaced by the appropriate values that correspond to the advanced aerodynamic knowledge that is presupposed by this figure.

For this type of engine nozzle protection, the differences in AOTV weight at end of mission are approximately the same (within 70 pounds) for 3, 4, 5, or 6 engine configurations. Consequently, this weight trade study suggests that 4 gimbaled or 6 hinged engines are preferable, but 3 or 5 gimbaled engines are also acceptable.

ENGINE NUMBER VEHICLE WEIGHT TRADE RESULTS

TRADE PERFORMED FOR 15' DIA LO_2/LH_2 OTV WITH SPHERICAL REAR OXIDIZER TANK & 15000 lb_f TOTAL THRUST

A - Δ AEROSHELL WEIGHT FROM PHASE I STUDY
B - Δ AEROSHELL WEIGHT IS NOT EFFECTED BY ENGINE NOZZLE LENGTH
C - Δ AEROSHELL WEIGHT BASED ON ADVANCED BASE HEATING TECHNOLOGY 30° NOZZLE PROTECTION ENVELOPE

	NUMBER OF ENGINES	1	2	3	4	5	6 HINGED	6 FIXED
I	BASELINE AOTV (LB)	14200	14200	14200	14200	14200	14200	14200
II	Σ OF PROPELLANT & PROPULSION SYSTEM (LB)	1592	1590	1505	1492	1562	1520	1500
A	Δ AEROSHELL WEIGHT OF PHASE I (LB)	393	373	213	140	6	0	0
B	Δ AEROSHELL WEIGHT WT OF AEROSHELL ≠ f (ENGINE LENGTH)	0	0	0	0	0	0	0
C	Δ AEROSHELL WEIGHT ADVANCED BASE HEATING KNOWLEDGE	0	160	55	0	0	0	0

NUMBER OF ENGINES		1	2	3	4	5	6 HINGED	6-FIXED
AEROSHELL CASE A I + II + A (LB)	W	16185	16163	15918	15832	15768	15720	15700
	ΔW	485	463	218	132	68	20	0
AEROSHELL CASE B I + II + B (LB)	W	15792	15790	15705	15692	15762	15720	15700
	ΔW	92	90	5	-8	62	20	0
AEROSHELL CASE C I + II + C (LB)	W	15792	15950	15760	15692	15762	15720	15700
	ΔW	92	250	60	-8	62	20	0

LOWEST WEIGHT

GRUMMAN

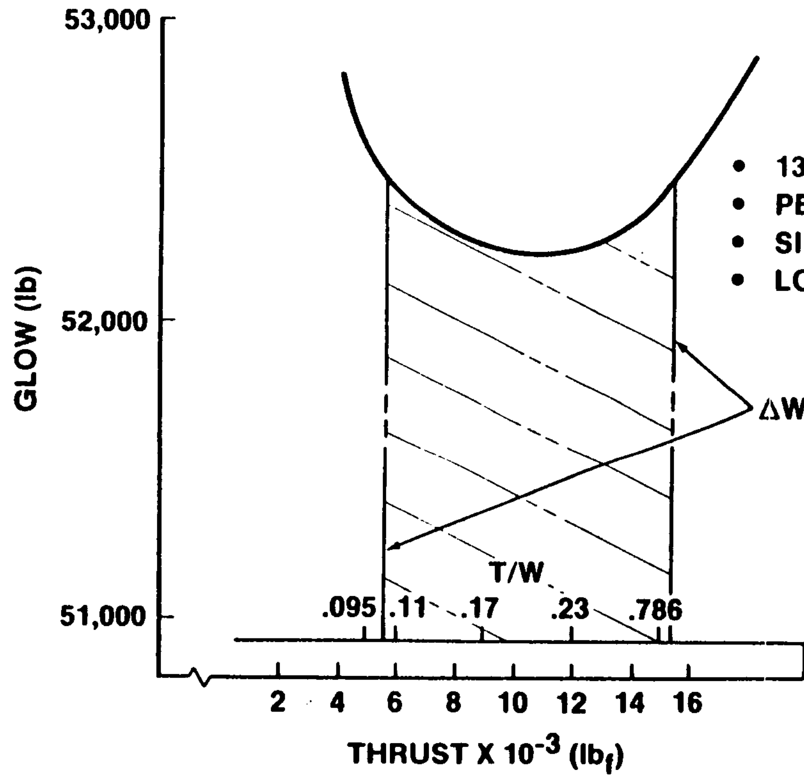
0283-013PP

A tradeoff study was performed to determine the optimum thrust level for an LO_2/LH_2 advanced engine small cargo carrying AOTV. The reference mission chosen was a 13,200 lb usable payload placement in GEO by perigee kick mode with a single perigee burn. Performance, size and weight data for the engines were provided by Aerojet. The same thermodynamic cycle was assumed over the thrust regime studied. A baseline test configuration was chosen and then fitted with a main propulsion system. By changing the engine(s) size(s), differing vehicle total thrust levels were obtained. The vehicle structure shell and propulsion system were resized for compatibility with the changing thrust level. The engine(s) performance was changed to account for the variation in specific impulse (I_{sp}) as a function of engine max thrust.

The facing figure illustrates a composite curve formed of a number of test cases with varying engine numbers. Each test case had a fixed engine number and the total thrust was varied to generate a characteristic curve. All the test case curves were virtually identical and were combined to generate the curve shown. The curve shown is therefore independent of engine number and is valid for aeroassisted delivery vehicles with dry weights of six to seven thousand pounds.

The optimized T/W chosen from this study was .15 to .25 which translates to a thrust requirement of 8000 to 13,000 lbf for the nominal payload delivery mission. A similar analysis on vehicles designed for the NASA reference 14K delivery and return manned GEO service mission suggests an optimum T/W of .15 to .25 when flown on a perigee kick delivery mission.

OPTIMIZED T/W FOR "NOMINAL" PAYLOAD DELIVERY



- 13,200 lbm USABLE GEO PAYLOAD
- PERIGEE KICK DELIVERY
- SINGLE PERIGEE BURN
- LO_2/LH_2 ADVANCED ENGINE

RECOMMENDATIONS FOR OPTIMIZED AOTVs WITH LO_2/LH_2:

- 8000 lb_f TO 13000 lb_f TOTAL THRUST
- FOR MAN-RATED VEHICLES, FOUR 3000 lb_f THRUST ENGINES

GRUMMAN

0283-014PP

3.3.5 Alternate Propellants

Although our primary emphasis in this study has been to explore technology variations on vehicles which use advanced liquid oxygen/liquid hydrogen engines for main propulsion, we have also looked into the implications of using other propellants. Our approach to the topic of alternate propellants is outlined in the figure.

We have brought to the AOTV communities attention, an exotic, not-yet-created material with exciting potential as a propellant for space vehicles: solid metastable helium (SMH). At the time of our presentation of this data, the Air Force was interested in supporting research on SMH.

We have also considered the implications of propellent density, as well as specific impulse, in selecting candidate propellents for an AOTV. The highest density propellant that we found that seemed attractive for AOTV use is Tetrafluorohydrazine/hydrazine.

ALTERNATE PROPELLANTS

- SOLID METASTABLE HELIUM
- COMPARISON OF AVAILABLE PROPELLANTS
  - DENSITY & I_{SP}
- TETRAFLUOROHYDRAZINE & HYDRAZINE

GRUMMAN

1590-003(T)

The characteristics of solid metastable helium, a possible material which has been considered for uses as an explosive and/or as a high energy propellant is outlined. At this time there are at least two unknowns which present serious consideration of MSH as a propellant for an advanced AOTV. These are:

- It is not known if MSH can be created.

- It is not known if its energy release can be controlled sufficiently to allow use within an engine (i.e., it may explode once it starts to give up energy).

Should these difficulties be overcome, MSH would offer 4 times the specific impulse of the highest known chemical propellants.

SOLID METASTABLE HELIUM

WHAT IS IT?

- SOLID MSH IS HELIUM WITH ITS ELECTRONS RAISED & MAINTAINED IN AN EXCITED STATE
- TRANSFORMATION FROM THIS STATE YIELDS SUBSTANTIAL ENERGY

DESIRABLE CHARACTERISTICS

- SOLID METAL AT ROOM TEMPERATURE
  - HIGH MELTING TEMPERATURE $\sim 570°F$
- VERY HIGH STORED ENERGY
  - THEORETICAL $I_{SP} = 3150$ SEC*
  - EXPECTED $I_{SP} > 2000$ SEC*
- USEFUL LIFE SPAN
  - EXPECTED STABLE FOR ~ 8 YEARS*

DEVELOPMENT STATUS

- THEORETICAL MATERIAL → NONE HAS BEEN MADE
  - CURRENT EFFORTS TO MANUFACTURE ARE (OR WILL SHORTLY BE) UNDERWAY AT JPL (DR J.S. ZMUIDZINAS)
- AIR FORCE FAVORS DEVELOPMENT
  - "A PREVIEW OF THE TECHNOLOGY REVOLUTION," GENERAL ROBERT MARSH, AIR FORCE MAGAZINE, AUGUST 1984
- *BACKGROUND SOURCE
  - "ALTERNATE PROPULSION ENERGY SOURCES," DR. R.L. FORWARD, AFRPL TR-83-067, DECEMBER 1983

GRUMMAN

1590-004(T)

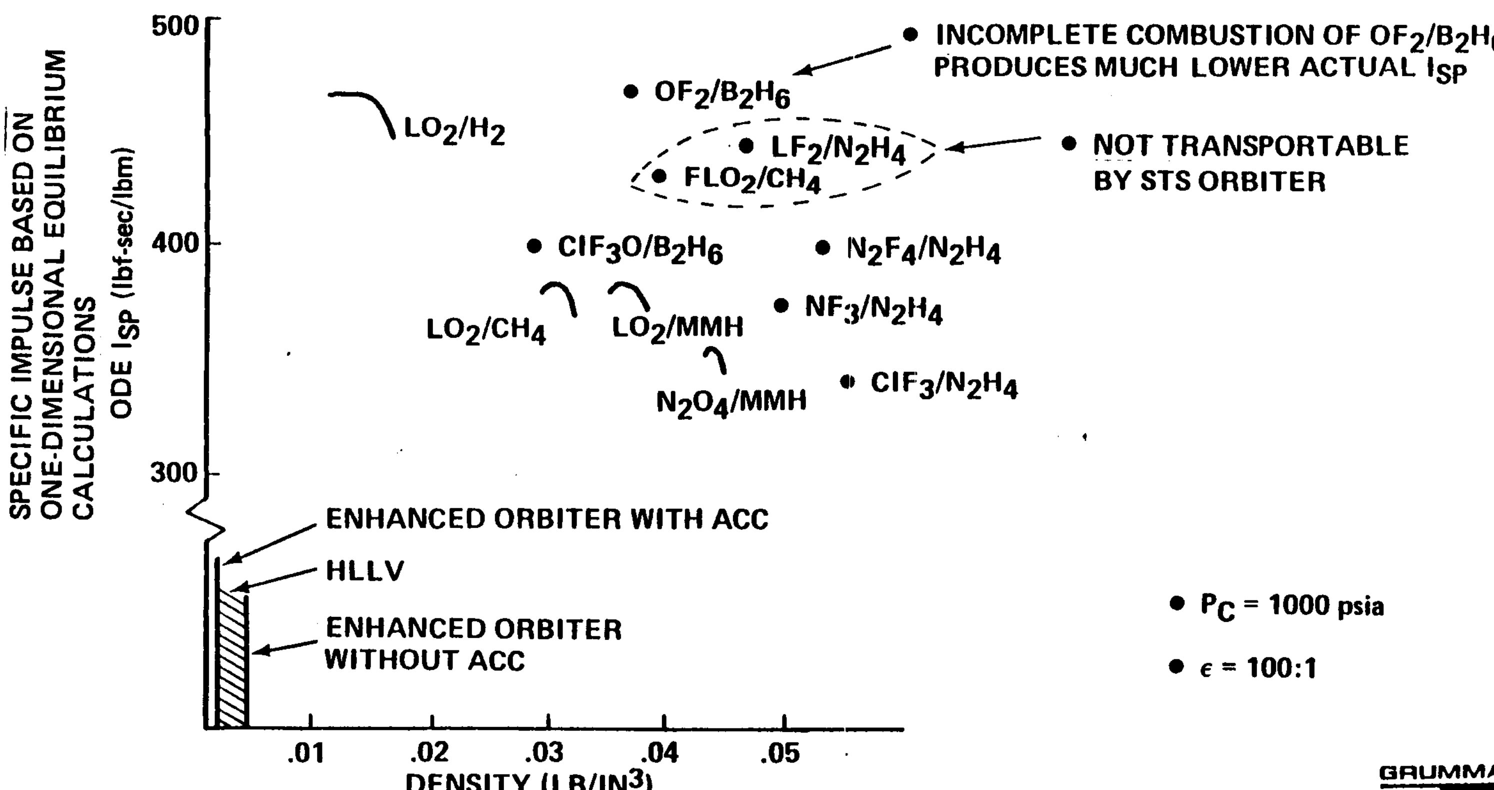
The facing figure illustrates the data used in evaluating alternate propellant choices of LO_2/LH_2. The bottom graph of the figure presents both the specific impulse and propellant combination density at the I_{sp} optimized mass ratio. The upper table lists two figures of merit for the ranking of propellant combinations. The specific impulse measures the impulse received from one pound of reactants. The density impulse measures the impulse received from one cubic foot of reactants. The propellant combinations are shown ranked left to right based on the density impulse. The specific impulse is widely considered the performance yardstick of propellant combinations, but consideration should be given to the density impulse performance because it heavily influences several aspects of AOTV program operational costs.

A high density impulse propellant combination has benefits to biconic AOTV designs. Less propellant tankage volume is required for a vehicle using a high density impulse propellant combination, and this decreases the size of the AOTV. The smaller AOTV size saves structural weight and TPS aeroshell weight. When compared to a larger vehicle, is more efficient. Ground based smaller vehicles also allow greater flexibility with the STS cargo bay length constraint. This allows the STS to better utilize its payload capacity. Based on the above, tetraflurohydrazine/hydrazine (N_2F_4/N_2H_4) was chosen as a propellant combination that is worth investigating as an alternate to LO_2/LH_2. N_2F_4/N_2H_4 is one of the propellant combinations that was chosen for concept evaluations in the Grumman computerized manifesting study. A discussion of this study can be found in a later section of this report.

COMPARISON OF AVAILABLE PROPELLANTS

PROPELLANT COMBINATION	N_2F_4/N_2H_4	LO_2/LH_2	N_2O_4/MMH	LO_2/MMH	LO_2/C_3H_8	LO_2/CH_4
*DENSITY IMPULSE lbf sec/ft^3	35203[2]	29308[1]	26788[1]	24043[2]	23594[2]	22949[2]
*SPECIFIC IMPULSE lbf sec/lbm	383[2]	480[1]	343[1]	373[2]	378[2]	381[2]

*AT E = 400/1 and I_{SP} MAXIMIZED MR
1. PERFORMANCE DATA SUPPLIED BY AEROJET
2. PERFORMANCE DATA SUPPLIED BY ROCKETDYNE



GRUMMAN

1590-005(T)

Some of the chemical, physical, and thermal properties of tetraflurohydrazine are listed in the figure. A summary of a material compatiblity study performed by Aerojet is also shown.

PROPERTIES OF TETRAFLUOROHYDRAZINE

• DATA SUPPLIED BY AEROJET

MOLECULAR FORMULA	N_2F_4
MOLECULAR WEIGHT	104.0 AMU
NORMAL BOILING POINT	−101.2°F
LIQUID DENSITY	103 LB/FT^3
GAS DENSITY	0.277 LB/FT^3 AT 70°F & 14.7 PSIA
VAPOR PRESSURE	6 PSIA AT -130°F
CORROSIVE CHARACTER	NONCORROSIVE
TOXICITY	HIGH, COMPARABLE TO N_2O_4 (HYDRAZINE IS RANKED "MEDIUM")
PHYSICAL FORM	COLORLESS LIQUID AND GAS

• DATA SUPPLIED BY AEROJET

MELTING POINT	−261.4°F
CRITICAL TEMPERATURE	96.8°F
CRITICAL PRESSURE	495 PSIA
HEAT OF FORMATION, LIQUID	−5200 CAL/MOLE AT 25°C
HEAT OF VAPORIZATION	3170 CAL/MOLE AT NBP
VISCOSITY, LIQUID	1.34 X 10^{-4} LB/FT SEC
THERMAL CONDUCTIVITY, LIQUID	8.7 X 10^{-2} BTU/FT HR °F AT −37°F

MATERIALS COMPATIBILITY INFORMATION FOR N_2F_4

• DATA SUPPLIED BY AEROJET

HARDWARE ITEM	SATISFACTORY FOR GASEOUS SERVICE	SATISFACTORY FOR LIQUID SERVICE
LINES, FITTINGS & STORAGE VESSELS	NICKEL, COPPER & CARBON STEEL POLYETHYLENE AND TYGON PYREX	STAINLESS STEEL, NICKEL COPPER PYREX
VALVE BODIES	STAINLESS STEEL, MONEL & BRASS (ZINC FREE) PYREX (STOPCOCKS)	STAINLESS STEEL
VALVE SEATS	STAINLESS STEEL & MONEL TEFLON*	STAINLESS STEEL TEFLON
VALVE PLUGS	MONEL	STAINLESS STEEL
VALVE PACKING	COPPER BRAID WITH TEFLON TEFLON AND KEL F	TEFLON
GASKETS	COPPER TEFLON	NONE USED
O RINGS	VITON A AND KEL F	NONE USED
SCREWED PIPE CONNECTIONS	TEFLON TAPE	TEFLON TAPE

*NYLON IS NOT SATISFACTORY AS A VALVE SEAT IN GASEOUS SERVICE

GRUMMAN

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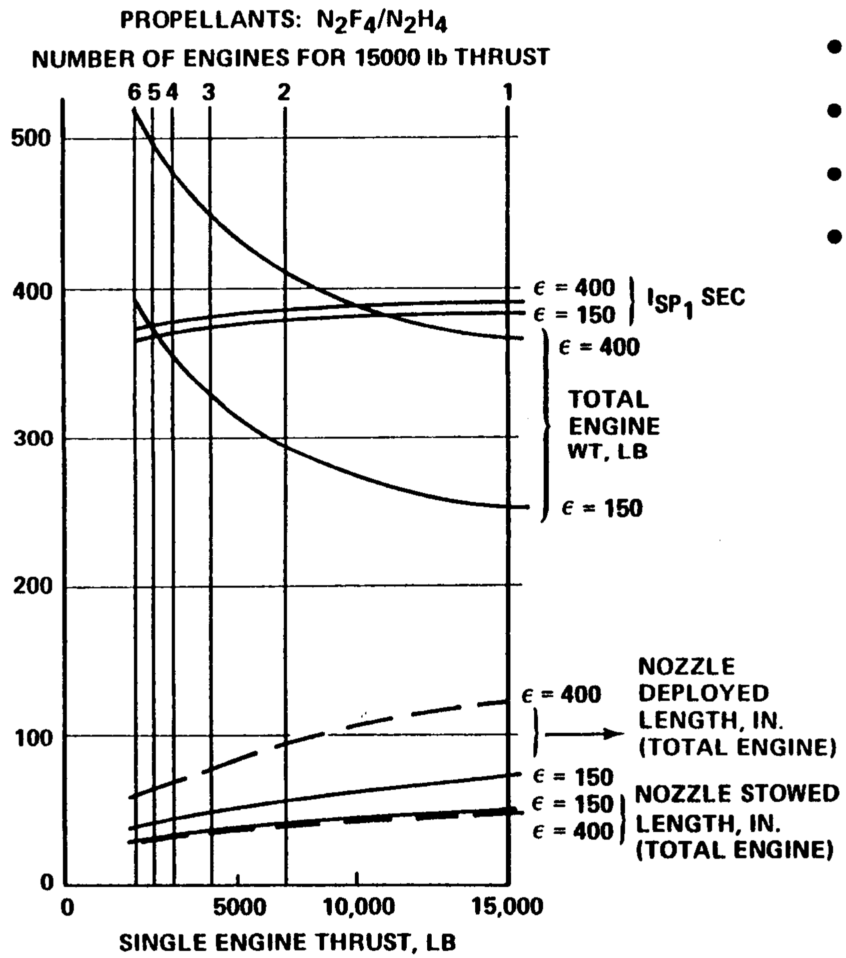
The figure on the facing page illustrates the trend data, derived from computer analysis performed by Aerojet Techsystems, on N_2F_4/N_2H_4 engines. It was used in the preliminary design of the N_2F_4/N_2H_4 modular vehicles presented in this report. Engine weight, length and performance are given as a function of single engine thrust level for systems with 15,000 lbf total thrust. A later rework of the analysis by Aerojet concluded that an 87% bell nozzle, run at a mixture ratio of 3.07/1, had superior performance to the data shown in the figure (which assumed a 115% bell nozzle run at MR = 2.76/1). The gain in specific impulse between the two nozzle types is approximately 6 lbf sec/lbm for the same thrust level. The higher I_{sp} nozzles are used in the tetraflurohydrazine/hydrazine vehicles evaluated in Grumman's computer manifesting program "Good Fit". The inputs and results of the computer tradeoff study are discussed in a later section.

TETRAFLUOROHYDRAZINE/HYDRAZINE ENGINE TREND DATA



- DATA SUPPLIED BY AEROJET
- ADVANCED TECHNOLOGY ENGINE
- ϵ = ENGINE NOZZLE EXPANSION RATIO
- PROPELLANTS
  - OXIDIZER = N_2F_4
  - FUEL = N_2H_4

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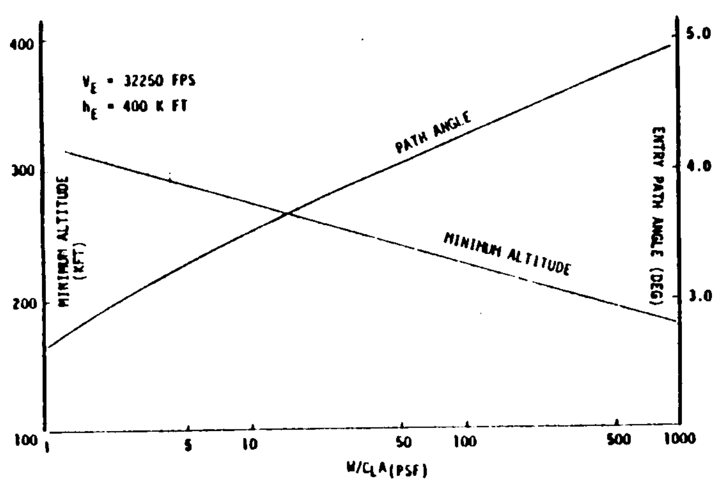
1590-008(T)

3.4 Thermal Protection Subsystem

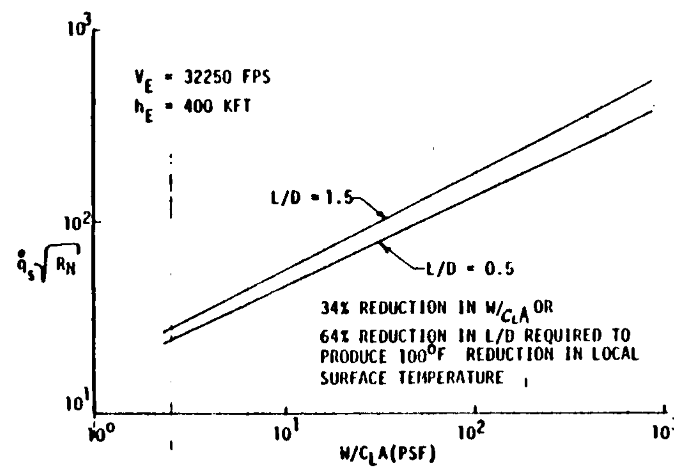
Modulation of the mid L/D AOTV lift vector is used to provide trajectory control (null accumulated errors) and to change the orbital plane (if desired). In previous sections, it was demonstrated that large performance gains are realized by placing the orbit inject propulsion on the payload, and utilizing the AOTV only as a reusable perigee kick vehicle. In this no-plane-change mode nearly all of the lift can be used to help "capture" in the one pass mode and thus the vehicle flies considerably higher, and hence cooler, than if the lift vector were being employed to change the inclination of the orbital plane. At the one pass overshoot bound, the minimum altitude, and hence maximum heat transfer, is driven only by the lift loading parameter W/C_LA. These variations are illustrated in the table and figure.

GEO RETURN OVERSHOOT BOUND

MINIMUM ALTITUDE



MAXIMUM HEAT TRANSFER



MINIMUM ALTITUDE OPTIONS FOR MID L/D AOTV

- L/D OF VEHICLE USED TO PROVIDE TRAJECTORY CONTROL (TO NULL ACCUMULATED ERRORS) AND PLANE CHANGE (IF DESIRED)

- GEO DELIVERY PERIGEE KICK VEHICLE REQUIRES NO PLANE CHANGE SO CAN OPERATE NEAR OVERSHOOT BOUND TO MINIMIZE MAXIMUM TPS SURFACE TEMPERATURES

- MANNED ROUND TRIP VEHICLE HAS LOWER OPERATIONAL COST WHEN L/D IS USED TO PROVIDE PLANE CHANGE (110M OVER 20 FLIGHTS) BUT OPERATES AT HIGHER TPS SURFACE TEMPERATURES

- AT ONE PASS OVERSHOOT BOUND, MINIMUM ALTITUDE IS DEPENDENT ONLY ON THE LIFT LOADING PARAMETER, W/C_LA

- PEAK HEAT TRANSFER RATE AT THE ONE PASS OVERSHOOT BOUND IS DEPENDENT BOTH ON W/C_LA (DRIVES MIN ALTITUDE) AND L/D (DRIVES VELOCITY DEPLETION)

	Vehicle	L/D	W/C_DA (psf)	W/C_LA (psf)	Stagnation Point $\dot{q}$ max BTU/FT^2 Sec	Return Flight Mode
(4)	13.2 Geo Delivery Perigee Kick + AKP	1.03	127	123	140	Overshoot Bound
(10)	14Klbs Up & Back (4) + crew module + Int Tank	1.2	525	438	266	Overshoot Bound
		"	"	"	615	Maximum Plane Change

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Boundary Layer Transition

An evaluation has been made of the possibility of boundary layer transition to turbulent flow employing our standard design approach. The boundary-layer transition criteria employed was a recently published Re_θ criteria. This correlation, applicable to carbonaceous-nosed vehicles with carbon-phenolic frusta, was developed during a recent evaluation of ballistic vehicle boundary-layer transition data sponsored by AFFDL. This correlation is presented in the figure on page and employs an $Re_{\theta t}$ vs X/R_N format where both onset and forward progression data are included. Local bounday-layer properties were generated by the 3VFF bounday-layer code, including nose and frustum mass addition, as well as boundayr-layer entrainment and wall temperature effects. The standard deviation of the local momentum thickness Reynolds number is 22% of the graphite/carbon-phenolic subset. A preliminary evaluation of transition onset and progression has been made of a maneuvering vehicle at angle of attack. Consistent with the ballistic vehicle correlation, local windward, leeward, and side ray boundary-layer propeties at transition were computed with the 3VFF code. The windward and side ray onset and progression data base agreed reasonably well with the ballistic vehicle. Hence, it appears reasonable to employ this correlation on the Mid L/D AOTV vehicle.

The variation of Re_θ with X/R_N for the Mid L/D AOTV at the time of peak heating is illustrated in the figure. Note that the windward meridian is predicted to be emersed in turbulent flow from about 9 Rns aft of the nose. Note also that if Re_θ could be reduced by about 40% by an increase in the minimum altitude, turbulent flow would not be predicted. It is expected that an increase in altitude or about 14,000 ft would provide the necessary Re_θ reduction to assure laminar flow.

In a recent survey of "Sensitivity of Thermal Protection System Design to Boundary Layer Transition Criteria" conducted by Nestler and Florence, numerous transition criteria were evaluated in the context of their impact on the Shuttle Derived Vehicle re-entry module. An alternate approach to that described above might be transition onset correlation no. 4 of that paper, as presented by Zoby and validated by data from the first shuttle flight. That correlation

$$Re_\theta/M_E = 275$$

essentially predicts laminar flow over the AOTV for the case illustrated, $Re_\theta/M_e = 120$, at 200 kft. It appears that an increase in Re_θ of about X2.3 could be tolerated before boundary layer transition would be predicted. This wold amount to a decrease of minimum flight altitude by about 23,000 ft.

Hence, it appears that a large uncertainty exists concerning the predicted onset of boundary layer transition to turbulent flow. Since the onset of turbulent flow would result in increased local heat transfer by about a factor of five, it appears mandatory to design the AOTV and mission scenario to maintain laminar flow. This will insure minimum weight and maximum reusable service life from the thermal protection system.

TWO CRITERIA APPLIED - CONTRADICTORY RESULTS OBTAINED

- BERKOWITZ ET AL Re_θ - AIAA PAPER #77-125
- SCOTTOLINE/ZOBY $Re_\theta/M_E = 275$ - AIAA PAPER #82-0002

$M_\infty = 30$ @ 200K FT RESULTS

- Re_θ PREDICTS TURBULENT FLOW 9 R_{NS} BACK ON W/W MERIDIAN, AN INCREASE IN MINIMUM ALTITUDE TO 214K FT WOULD ASSURE LAMINAR FLOW SO FLYING AT OVERSHOOT BOUND (227K FT) WOULD RESULT IN ALL LAMINAR FLOW FOR THE DELIVERY VEHICLE.

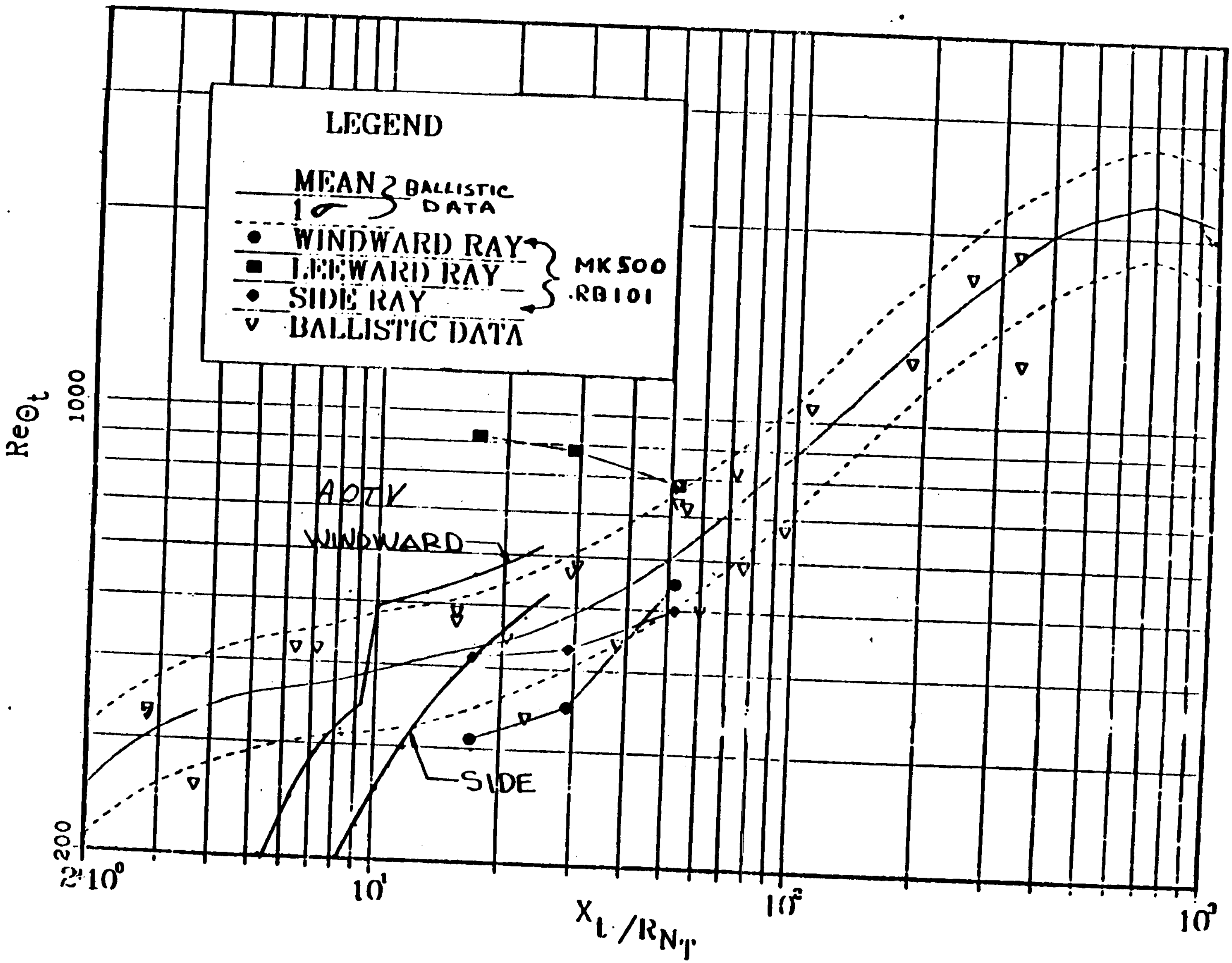
- THE 14K ROUND TRIP VEHICLE WOULD BE AT 197K FT FOR THE OVERSHOOT BOUNDARY AND THUS EXPECT TO EXPERIENCE SOME TURBULENT FLOW BY THIS CRITERIA.

- Re_θ/Me PREDICTS LAMINAR FLOW TO MINIMUM ALTITUDES DOWN TO 177K FT, BOTH THE DELIVERY AND THE 14K ROUND TRIP VEHICLES FLYING ON THE OVERSHOOT BOUNDS ARE EXPECTED TO BE ALL LAMINAR FLOW.

IF TURBULENT FLOW EXISTS ⟶ TPS WEIGHT PENALTY, AVOID TURBULENT FLOW BY FLYING NEAR OVERSHOOT BOUND ⟶ COSTS MAIN PROPELLANT PENALTY

See discussion on previous page.

FRUSTUM TRANSITION CORRELATION FOR VEHICLES
HAVING GRAPHITIC NOSES & CARBON PHENOLIC FRUSTA



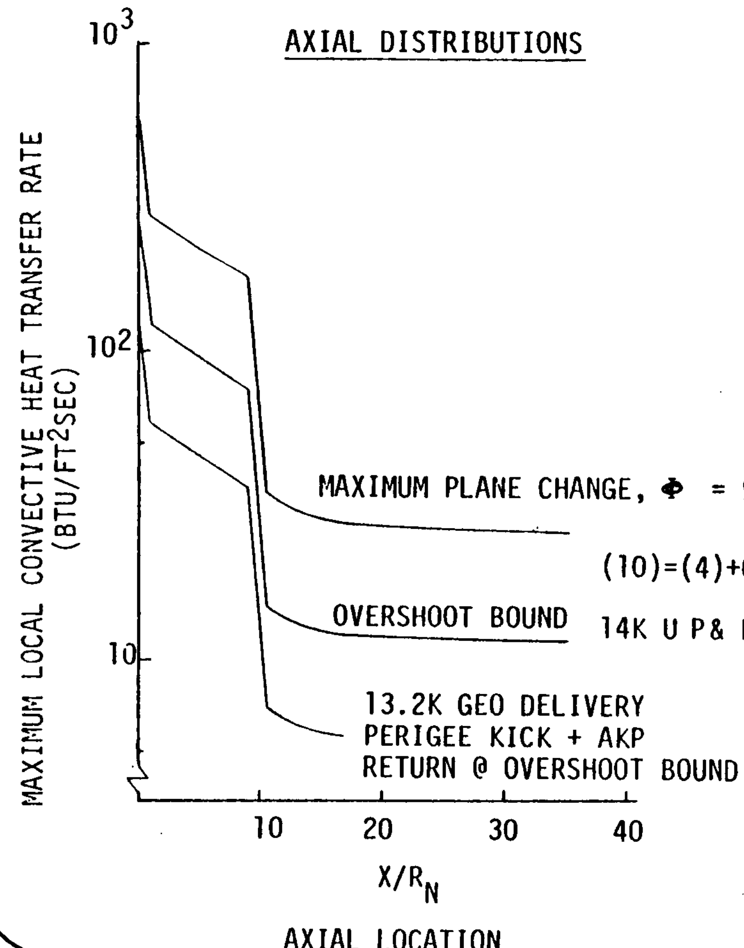
Local hypersonic heat transfer to a fully catalytic surface at several angles of attack (α) has been computed employing the 3D viscous boundary layer code (3VFF). Axial distributions along the windward meridian at $\alpha = 15^{o}$ for several missions are illustrated. The circumferential distribution at the end of the aft frustum is also illustrated. Note from the wide differences of the axial variations, a unique delivery vehicle would have a very different TPS than a manned vehicle were the lift force generated is employed to change the orbital plane. Note also that the AOTV sides and leeward areas will have very different TPS than the windward side.

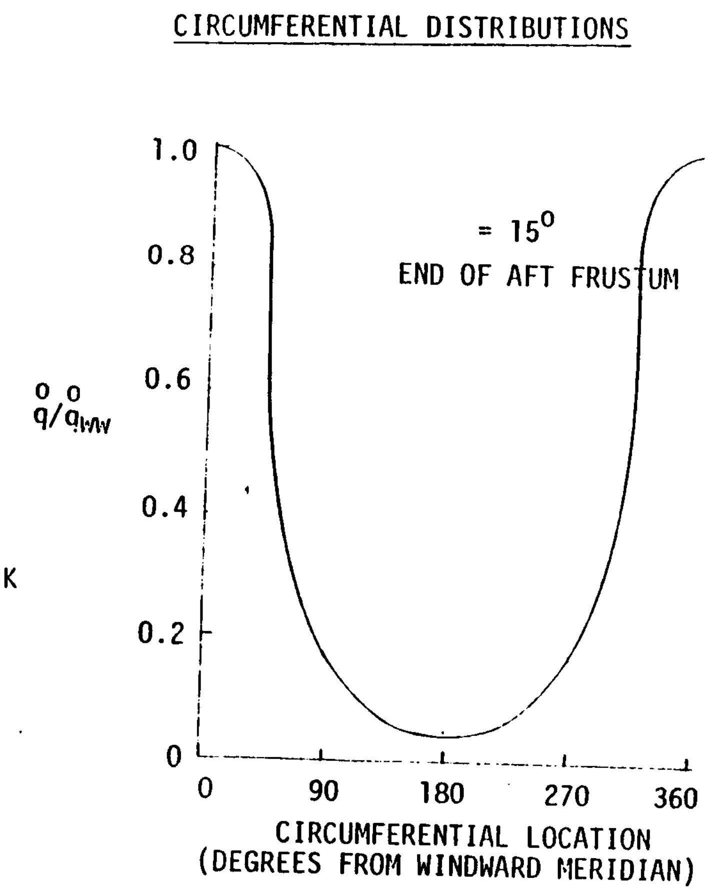
Accuracy of this hypersonic heat transfer distribution prediction should be evaluated by exercise of 3VFF for ground tests recently conducted on this class of vehicles.

C-4

HYPERSONIC CONVECTIVE HEAT TRANSFER TO AOTV'S

- o COMPUTATIONS PERFORMED AT M_∞ = 30, h = 200 KFT, FULLY CATALYTIC SURFACE
- o GE 3D VISCOUS BOUNDARY LAYER CODE (3VFF) EMPLOYED (AFFDL-TR-78-67)
- o AXIAL AND CIRCUMFERENTIAL HEATING DISTRIBUTIONS AVAILABLE
- o BOUNDARY LAYER EDGE CONDITIONS, SHOCK GEOMETRY, AND OTHER REFERENCE PARAMETERS AVAILABLE





Using the heat transfer distributions described on the previous page and the parametric trajectory results previously reported, local maximum heat transfer rate distributions for the windward meridian have been generated for several AOTV missions for a fully catalytic surface. If a high temperature coating could be developed that exhibited a totally non-catalytic surface, it is expected that the local heat transfer would be reduced by a factor of about 3. Employing a surface emittance of 0.8, radiation equibibrium surface temperatures have been generated, Page for both fully catalytic and non-catalytic surfaces.

EFFECT OF MISSION & SURFACE COATING CHARACTERISTICS ON MAXIMUM TPS SURFACE TEMPERATURE

- MAXIMUM SURFACE TEMPERATURES GENERATED FOR RADIATION EQUILIBRIUM, $\epsilon = 0.8$, AND BOTH FULLY CATALYTIC AND FULLY NON-CATALYTIC SURFACES
- PUBLISHED AOTV COMPUTATIONS BY SCOTT SUGGEST $\frac{\dot{q}_{non\ cat}}{\dot{q}_{cat}} = 0.3$ (AIAA-84-1710)
- LARGE EFFECT ON MAXIMUM SURFACE TEMPERATURES PRODUCED BY
  - VEHICLE SURFACE LOCATION
  - TOTALLY NON-CATALYTIC COATING
  - STEERING LAW EMPLOYED (OVERSHOOT BOUND VS PLANE CHANGE)
  - MASS OF RETURN VEHICLE (EMPTY DELIVERY VEHICLE VS MANNED RETURN)
  - ANGLE OF ATTACK

See discussion on previous page.

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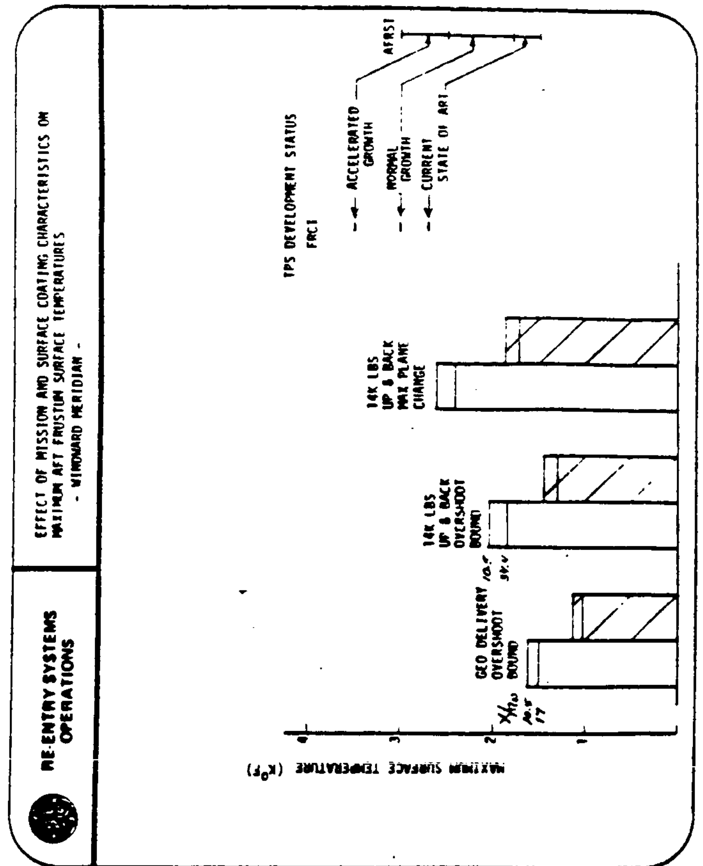
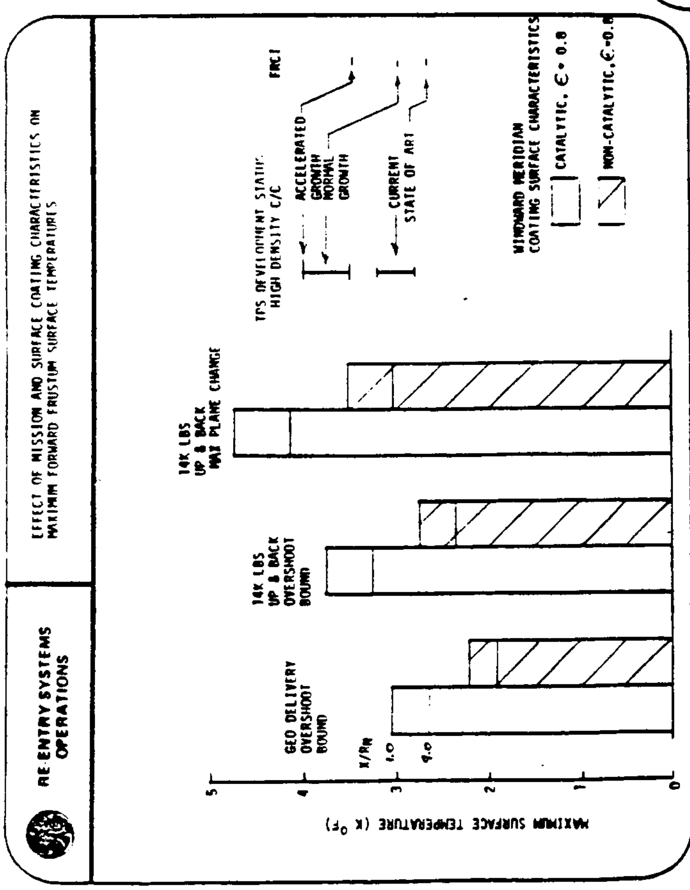




As a part of the Thermal Protection Subsystem Trades Task and at the suggestion of Bob Jackson, LaRC, a review has been made of the recent NASA contracted activity on heat pipes for Space Shuttle leading edge applications. The results on design, fabrication and testing of a sodium driven Hastelloy-X heat pipe have been described in References 1, 2 and 3. The predicted operating temperature, given a maximum heat transfer rate, is shown in the figure. Also illustrated is the hot wall heat flux-maximum temperature experienced in the Mach-7 hypersonic flow test. An additional higher temperature heat pipe capability for a coated Columbrium heat pipe is illustrated (by extrapolation of the Hastelloy-X performance trend) by the dashed line. It appears that the demonstrated and projected heat pipe designs offer compatitive maximum use temperature capability with current and growth reusable TPS, and based on the result summarized in Reference 1, are also weight competitive with the RCC. Therefore, areas exist on most of the AOTV configurations where use of heat pipes would be expected to be a viable option.

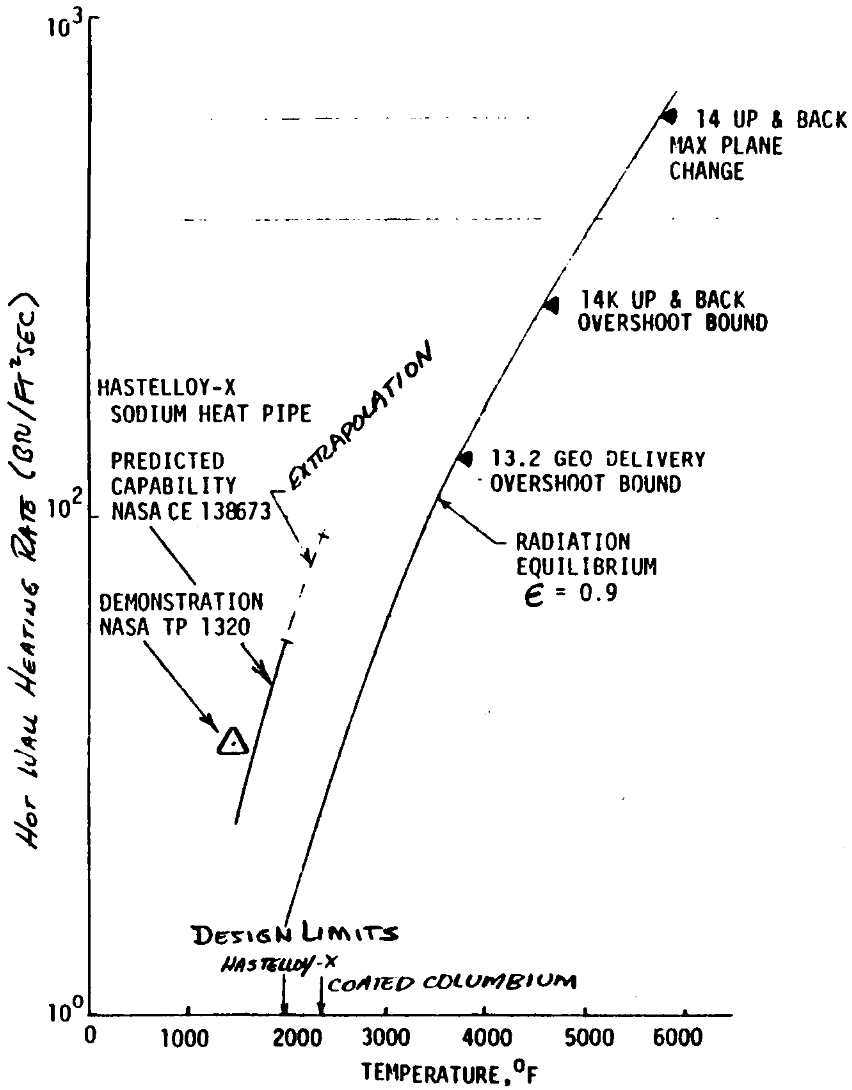
References

1. Niblock, G.A., J.C. Reeder, And F. Huneidi, "Four Space Shuttle Wing Leading Edge Concepts", JSR May 1974, Vol II, No. 5, page 314.

2. Anon, "Design, Fabrication, Testing, and Delivery of Shuttle Heat Pipe Loadng Edge Test Modules", NASA-CR-138673, 20 April 1973.

3. Camarda, C.J., "Aerothermal Tets of a Heat-Pipe-Cooled Loading Edge at Mach 7", NASA-TP-1320, November 1978.

UTILITY OF HEAT PIPES FOR MID L/D AOTV



- HEAT PIPES OFFER LIMITED UTILITY FOR MID L/D AOTV NOSES

- APPLICABLE TO FORWARD FRUSTUM AREA

3.5 Structure Subsystem

In sizing the structure for a space based AOTV it is seen that the design conditions for a space based vehicle are much less severe than for a ground based vehicle. The ground based AOTV must withstand the Orbiter launch accelerations while fully loaded with propellant. The space based vehicle in operation is only required to withstand the engine thrust loads and the aerodynamic pressures during re-entry.

Ultimate load factors for probable critical conditions are shown in the table. A 62,000 lb GLOW manned AOTV was selected for the trade study.

Typical shell analysis methods were used to give design loads for comparison.

EFFECT OF SPACE BASING ON MID L/D AOTV STRUCTURAL MASS REQUIRED

ASSUMPTIONS:

- VEHICLE LOADING DURING LAUNCH IN ORBITER IS MORE CRITICAL THAN THAT DURING ATMOSPHERE RE-ENTRY.

AXIS	ULT LOAD FACTOR GROUND BASED	ULT LOAD FACTOR SPACE BASED
N_X	6.75	1.97
N_Z	±4.50	±3.75
N_Y	0	0

AOTV $L = 53'$ $R_B = 7.5'$

MANNED CAPSULE + P/L DEL. = 6K#
MASS IN ORBITER = 62K#
MASS AT AEROMANEUVER = 11K#

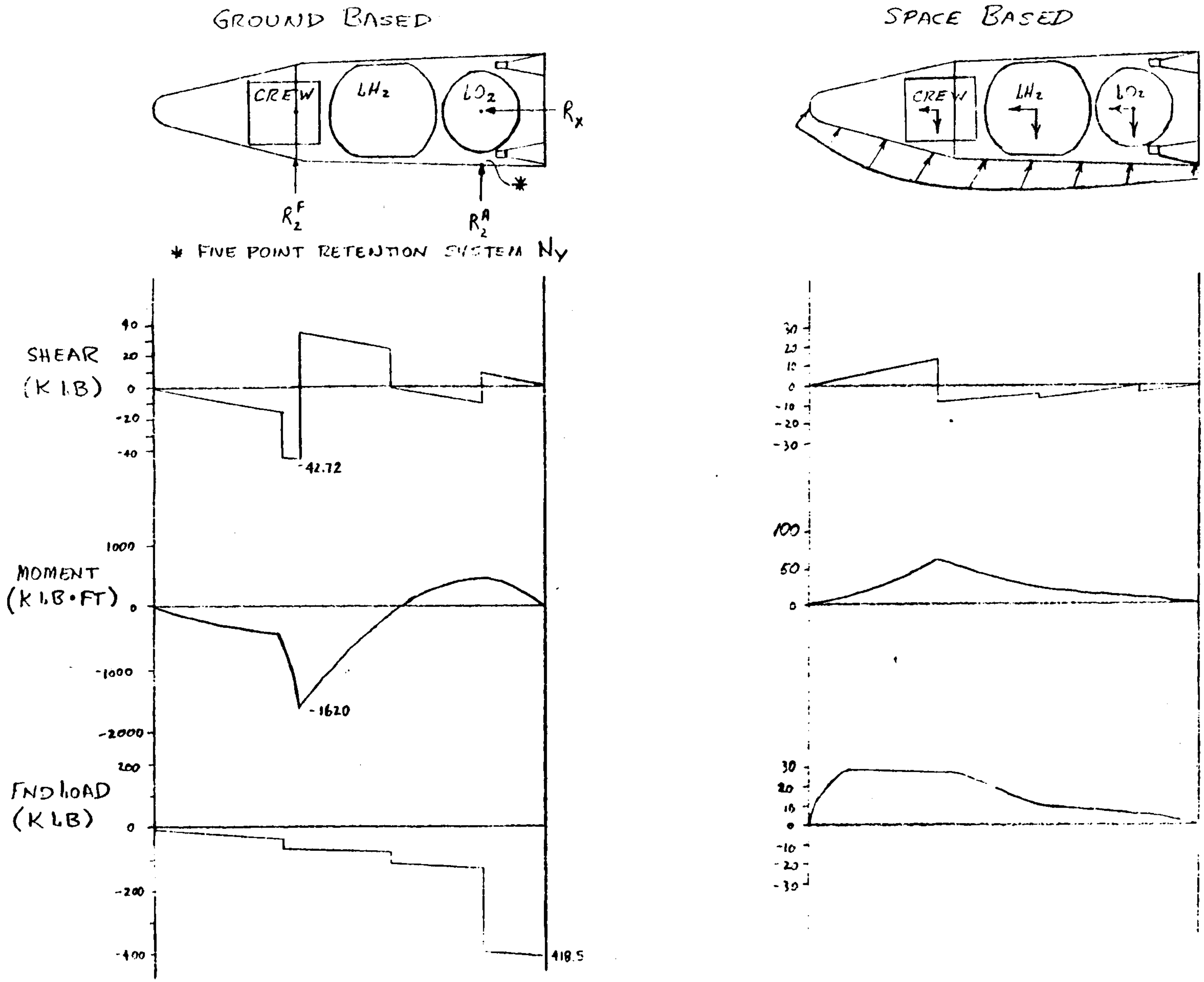
ANALYSIS CONDUCTED:

- BALANCED VEHICLE LOADS
- CONSTRUCTED SHEAR, MOMENT AND END LOAD DIAGRAMS
- SOLVED FOR STRUCTURE LOADING AT TWO STATIONS
- COMPARED STRUCTURE MASSES TO WITHSTAND LOADINGS

In the analysis of the ground based case the load factors were applied to the vehicle masses and reacted by the Orbiters "five point" suspension system as used by NASA. Overall shears, moments and end load diagrams are shown.

The acceleration loads for the AOTV during re-entry result from a distributed air pressure on the shell. This distributed reaction minimizes the shear, moment and end load compared to the concentrated reactions in the Orbiter. Of course, the load factors are also less.

SHEAR, MOMENT & END LOAD DIAGRAMS FOR MID L/D AOTVs



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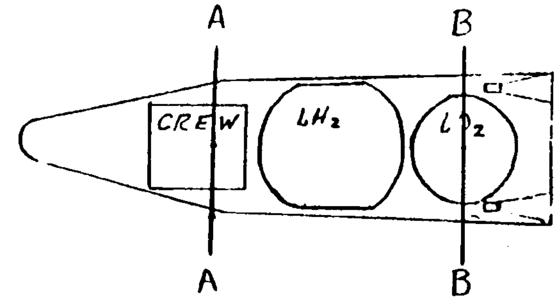
For shell analysis a load summary was made at the two stations of the AOTV where the bending moment was maximum; Section A-A at the forward Orbiter support and Section B-B at the aft support. The shear load is carried in the shell skin. The moment and end loads are combined to give tension/compression design loads for the shell in pounds per inch of shell periphery. For the space based shell the air pressure loads on the curved surface are not deemed critical for design.

For frames and local equipment support structure a ratio of combined load factors makes the best comparison.

LOAD SUMMARY FOR MID L/D AOTV



SHELL

SECTION	SHEAR GB	SHEAR SB	MOMENT (K#.FT) GB	MOMENT (K#.FT) SB	END LOAD (K#) GB	END LOAD (K#) SB	UNIT SHELL LOAD(#/IN) GB	UNIT SHELL LOAD(#/IN) SB
A-A	47.7	13.2	1620	65.5	68.6	25.8	185	49.5
B-B	14.5	3.4	399	11	406	4.6	735	8.6

FRAMES & EQUIPMENT SUPPORTS

RATIO OF LOAD FACTORS

$$\text{RATIO SB/GB} = \frac{\dfrac{N_Z^{SB}}{N_Z^{GB}} + \dfrac{N_X^{SB}}{N_X^{GB}}}{2} = \frac{\dfrac{3.75}{4.50} + \dfrac{1.97}{6.75}}{2} = 0.56$$

For the shell structure that withstands moment and end loads a ratioing of the unit loads is used, ie., for section A-A the ratio is SB Unit End Load/GB Unit End Load (49.5/18.5 = 0.27). However, the shell must support the TPS, maintain contour for air loads and provide micrometeorite protection so a shell and frame mass reduction ratio for the SB AOTV compared to the GB AOTV of 0.6 is considered achievable.

AOTV tankage structure has been assumed equivalent for the GB and SB cases due to pressure critical design criteria. However, if space operations can be done with lower pressures this ratio can be reduced.

Flaps are used only during re-entry and the loads will be the same for SB and GB.

SUMMARY OF MID L/D AOTV

SPACE VS GROUND BASED SUBSYSTEM MASS REQUIREMENTS

SUBSYSTEM	POTENTIAL SPACE/GROUND BASED MASS RATIO	NOTES	MASS RATIO
SHELL STR.	0.012 TO 0.27	REDUCTION IN SHELL STRUCTURE MASSES ARE LIMITED BY: TPS SUPPORT AEROELASTICITY MICROMETEORITE PROTECTION	0.6
FRAMES & EQUIP.SUPPORTS	0.56	FACTOR BASED ON RESULTANT LOAD FACTOR $f = \dfrac{\dfrac{N_Z^{SB}}{N_Z^{GB}} + \dfrac{N_X^{GB}}{N_X^{GB}}}{2}$	0.6
TANKS	1.0	LOWER PRESSURE REQUIREMENTS FOR SPACE OPERATION MAY REDUCE THIS RATIO	1.0
FLAPS	1.0	FLAP LOADS ARE NOT CHANGED BY SPACE BASING	1.0

While considering the implications of basing an AOTV at Space Station, a major concern that surfaced was the ability of Space Station personnel to adequately inspect an AOTV both after a flight and prior to a flight, so that the structural integrity of the AOTV can be assured before the start of a mission. To deal with this issue, we have offered the positive assessment which appears in the next few figures of a lightweight, automatic damage warning system which utilizes acoustic emissions from the AOTV structure as a monitored parameter.

Another space basing topic which follows the acoustic emission discussion is a discussion of approaches for designing and fabricating low cost external (supplementary) tankage for AOTV.

SOME SPACE BASING IMPLICATIONS

- STRUCTURAL HEALTH MONITORING
- LOW-COST DROP TANKS
  - CONVENTIONAL TOROIDS & ELLIPSOIDS
  - ROLL-UP CYLINDRICAL TANKS
- LIGHT WEIGHT VEHICLE STRUCTURES - THIRD STUDY PERIOD

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1590-022(T)

The facing page outlines some of the characteristics of an acoustic emission (AE) system for use in a biconic AOTV. The system is composed of an array of lightweight sensors, located at known places on selected structural members. By mathematically manipulating the signals received from several sensors (i.e., triangulation), a zone which contains a damaged member is identified. This operation greatly reduces astronaut inspection time for locating damage, and eliminates the need for routine, survey type physical inspections of structure between flights.

The system can have the ability to discriminate among the sounds it hears, and to only process sounds that are known to occur when physical damage occurs:

- Yielding
- Tearing
- Crack Growth

IMPACT DAMAGE DETECTION USING ACOUSTIC EMISSION (AE)

- CAN LOCATE POINT OF IMPACT
- FOR SHIELDED SHELLS, CAN LOCATE IMPACT ON INNER SHELL
- NO SCANNING REQUIRED
- DETECTS YIELDING, TEARING, OR CRACK GROWTH UNDER LOAD
- SYSTEMS AVAILABLE THAT CAN IGNORE SPECIFIC SIGNALS
  - TOO SMALL TO BE IMPORTANT
  - FROM NONCRITICAL AREAS
- APPLICABLE TO METALS AND COMPOSITE MATERIALS

GRUMMAN

1590-023(T)

The Figure highlights some current usage of AE technology in the aerospace field. Flying AE systems are in use in Canada and Australia. At Grumman, we have used AE in static tests on aluminum and composite structures.

PRESENT IN-FLIGHT SYSTEMS IN AIRCRAFT

- CRITICAL LOCATIONS MONITORED
  - WING
  - FUSELAGE
- PICKUPS ARE LIGHTWEIGHT
- RECORDERS – READ INTERMITTENTLY – RESULTS COMPUTED
- ALARMS PRACTICAL
- AUSTRALIA & CANADA MONITOR CRITICAL LOCATIONS ON AIRCRAFT USING AE
- USAF CONTRACT TO GAC FOR USE OF AE IN STATIC TESTS ON AIRCRAFT
- GRUMMAN HAS USED AE ON COMPOSITE STRUCTURES STATIC TEST TO PREDICT FAILURE LOAD ACCURATELY

GRUMMAN

1590-024(T)

The Figure lists some new technology efforts that will be helpful to the use of AE on AOTV. Some of the damage which may occur to an AOTV results from impact with particles of various sizes at hypervelocities (e.g., 15,000 ft/sec relative velocity). Since the acoustic signature of these types of impacts are presently unknown, some technology development work will have to be done in this area.

Another way of determining the location of a structural fracture (crack or tear) in a critical component is suggested in the Figure. It utilizes a grid of orthogonal fiber optic bundles with a light source at one end and a light sensor at the other end of each fiber bundle. When the structure tears and moves, it ruptures the fiber optic bundles in the area, interrupting their light signals. Since this occurs in two perpendicular directions, the location of the damage is known.

NEW TECHNOLOGY NEEDED FOR IMPACT DAMAGE DETECTION

- USE ACOUSTIC EMISSION (AE) TO DETECT BALLISTIC DAMAGE
- EFFECT OF SIMULATED MICROMETEROID IMPACTS USING AE DESIRABLE
- LOCATE TEAR IN LARGE STRUCTURE WITH FIBER-OPTICS GRIDS EMBEDDED IN COMPOSITES OR BONDED ON INTERNAL FACE OF METAL SHEET
- USE OF LINEAR AND PHASED ARRAYS OF ULTRASONIC DETECTORS TO MONITOR REGION IDENTIFIED AS DAMAGED
- MINIATURIZED ON-BOARD COMPUTERS FOR REDUCTION OF AE DATA

GRUMMAN

1590-025(T)

In general, auxiliary external main propulsion tankage has applicability to any AOTV. For biconics, it allows a significant increase in mission capability even though the normal AOTV tanks are confined within the fixed envelope of the aeroshell. Auxiliary external tankage can be used in one of two ways:

- Reuseable - which required the AOTV to fly an all propulsive mission and return the tanks to the point of origin. This may cost a substantial amount for transport of extra propellants.

- Non-Reused Drop Tanks - which are placed into a burn up orbit prior to the AOTV's atmospheric entry. This cost may be excessive unless inexpensive drop tanks are developed.

The subject of the Figure is low cost tankage, either reuseable or non-reuseable. It outlines an existing technology which has produced a significant number of bulge-formed tanks. However, all have been substantially smaller (e.g., less than 3' diameter) than AOTV would need.

LOW-COST DISPOSABLE TANKAGE BY BULGING PREFORMS

- PREFORM = UNDERSIZED VESSEL TO BE EXPANDED TO FINAL SIZE BY PRESSURIZATION
- PREFORMS MADE FROM LOW-COST DEVELOPABLE SHAPES WELDED TOGETHER
- OMIT WELD FIXTURING TO ELIMINATE WELD CONTRACTION
- AT LOW COST, BULGING WILL:
  - PRODUCE DOUBLE CURVATURE
  - ELIMINATE WELD INDUCED DISTORTIONS
  - WORK-HARDEN THE MATERIAL
  - DEMONSTRATE RELIABILITY OF FABRICATION

GRUMMAN

1590-026(T)

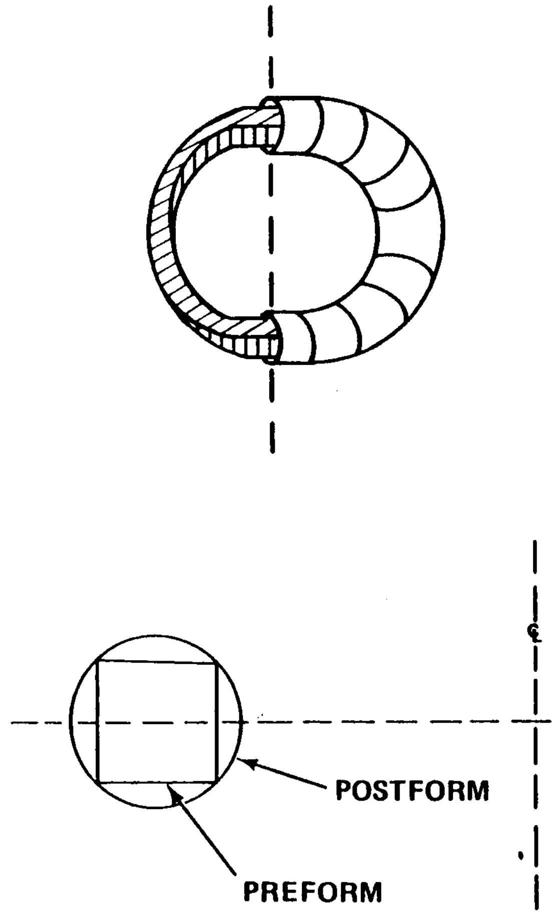
The Figure depicts the process of bulge-forming. The right side of the figure shows a cylindrical preform (upper left part) which becomes an ellipsoid (upper right part). The process is shown schematically in the lower right part of the figure. Internal pressure at cryogenic temperatures causes the preform to yield substantially as it becomes ellipsoidal. A similar schematic is shown (the left side) for forming a torroidal tank.

The process is low cost for a number of reasons:

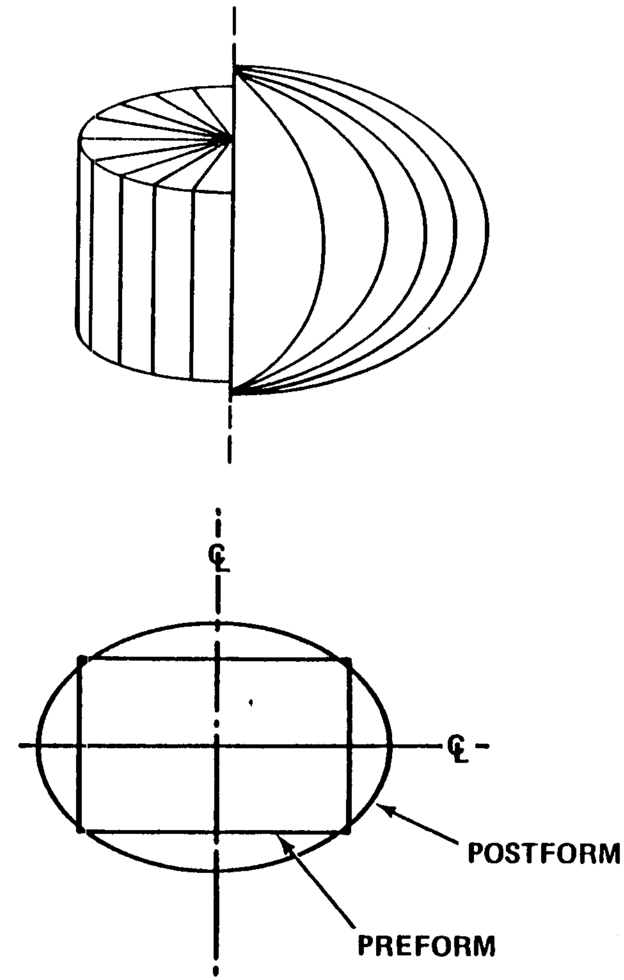
- The preform is very inexpensively made. For a cylinder, two flat sheets cut into a circle (the 2 ends) are welded to an open cylinder, which was formed by a single rectangular sheet whose opposing sides were welded together. Weld fixturing is not needed.

- Post-forming proof of strength tests are not required, since the internal pressure which yields the preform during the cryo-forming process exceeds the user pressures, and the vessel has survived this higher pressure.

TANK BULGING: PREFORMS TO POSTFORMS



TOROID



ELLIPSOID

GRUMMAN

1590-027(T)

The Figure outlines some of the material characteristics needed for bulge-formed tankage. It also lists some additional reasons why this is a low cost fabrication process, and, why the resulting tank is structurally efficient.

MATERIAL SELECTION CONSIDERATIONS

- COMPATIBLE WITH PROPELLANT
- AVAILABILITY (SHEET, FORGINGS)
- HIGH STRAIN TO NECKING PERMITS SIMPLER PREFORMS
- 100% WELD EFFICIENCY ELIMINATES PROVISION FOR DOUBLE THICKNESS JOINTS
- SIMPLE ANNEALING, NO QUENCH REQUIRED
- HIGH STRENGTH/WEIGHT PRODUCED BY WORK-HARDENING

GRUMMAN

1590-028(T)

The Figure lists some other factors which influence the cost or structural performance of these tanks. The last item in the figure refers to existing government-owned facilities which might be used to fabricate large size tanks with this methodology.

BULGING ECONOMIC OPTIONS

- STRESS CONCENTRATIONS AT DISCONTINUTIES & WELD ELIMINATED BY SIZING AT ROOM TEMPERATURE FOLLOWED BY ANNEAL
- CRYOBULGE
  - INCREASE DUCTILITY
  - INCREASE SUBSEQUENT RT STRENGTH
- RESIDUAL CORRUGATIONS PERMITTED IN REGION FREE OF ATTACHMENTS
  - TO SIMPLIFY PREFORM
- AVAILABILITY OF FACILITIES
  - MISSISSIPPI TEST STAND } NEEDED FOR VERY LARGE DROP TANKS
  - SATURN TANK AS CRYOSTAT } NEEDED FOR VERY LARGE DROP TANKS
  - LN_2 BARGES FOR CRYOGEN AT PRESSURE

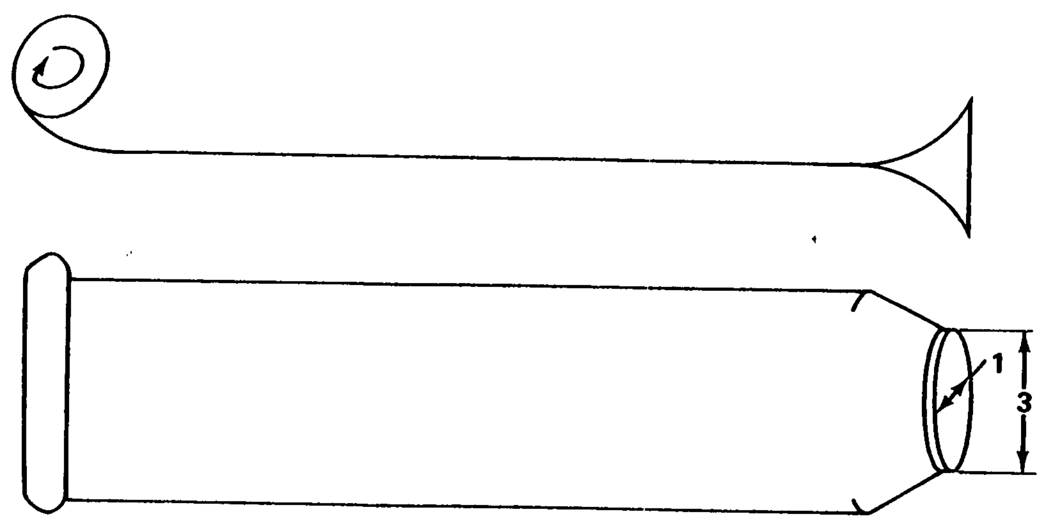
GRUMMAN

1590-029(T)

Some scenarios for Space Station as a transportation node envision the accumulation and storage of large quantities of propellants at Space Station. One approach to auxiliary tankage for AOTV is to bring empty tanks to Space Station, where they would be filled with propellant for an AOTV mission. This approach will be most cost efficient if the empty tanks do not occupy a large volume of space within the earth-LEO transport vehicle (e.g., STS Orbiter). One technique of doing this is shown. In a manner analogous to a used tube of toothpaste, a thin walled tank is wound up around a drum. To allow for rigid attachments for plumbing and structurally acceptable deformations, a rigid fixed end of elliptical shape (and 3:1 proportions) transitions gradually into the circular cross-section of the rest of the cylinder. This transition zone is the principal source of STS cargo bay length occupied by the tank in its rolled up configuration. In the cargo bay, a large tank will utilize about 10 feet of cargo bay length. This is substantially shorter than the 32′ length shown in Section 3.2.2.3.

ROLL-UP DISPOSABLE CYLINDRICAL PRESSURE VESSELS

- ULTRATHIN WALL STRUCTURES
- TOOTHPASTE TUBE-TYPE ROLL-UP PROBLEMS
  - FLATTENING TUBE CAN PRODUCE LARGE BENDING STRAINS (20%) IF BEND RADIUS = 4 THICKNESS
  - SUBSEQUENT BENDING STRAIN OVER A 12-IN. RADIUS PRODUCES SMALL STRAINS AT RIGHT ANGLES
  - THE INNER FACE WILL BUCKLE ELASTICALLY
  - TO REDUCE SHUTTLE LENGTH REQUIRED, THE HELD END SHOULD BE AN ELLISPOID WITH 3:1 WIDTH TO HEIGHT RATIO
  - TO MINIMIZE HIGH STRESS CORRUGATIONS ON PRESSURIZATION, AN ADEQUATE END SHAPE WILL HAVE TO BE DETERMINED



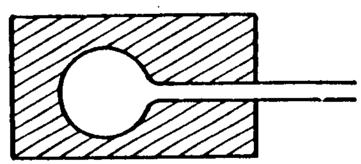
GRUMMAN

1590-030(T)

One of the problems with this type of tank is the possibility of forming a crease or kink from doubly curved surfaces. The method outlined on the Figure precludes this at the edges of the empty, flat tank by providing edge guides on the wind up drum which controls the shape the tank takes as it is wound up. The grooves at the edge of the drum will provide a continuous lateral load path for the wound up tank. This will prevent sliding and binding of the rolled up drum during launch vibrations. By providing a controlled, predictable path for the rolled up tank, the drum edge grooves provide designers with a method of predicting, and limiting, tank stress during roll up.

SOME POSSIBLE SOLUTIONS FOR ROLL-UP TANK PROBLEMS

- ROLL-UP WITHOUT KINKING
  - USE A SPIRAL-SHAPED GUIDE AT THE EDGES OF THE DRUM. THIS APPLIES BENDING GRADUALLY AND UNDER TENSION
- LAY-UP OF MULTIPLE ROUND SURFACES
  - AVOID LATERAL SLIDING AND INDETERMINACY BY EITHER EDGE COVERS



  OR SPACERS NEAR EDGES WHICH PROVIDE FLAT SURFACES AND WHICH SEPARATE THE ROUNDED SURFACES FROM EACH OTHER

- BENDING DURING ROLL-UP
  - THE BENDING STRAINS ARE PROPORTIONAL TO VESSEL THICKNESS:
    - $\epsilon \approx 0.01$ FOR $t = 0.040$ IN.
    - $\epsilon = .005$ AT $t = .020$ IN., YIELDING WILL PROBABLY OCCUR DUE TO BIAXIAL EFFECT (i.e., BECAUSE OF PRIOR YIELDING AT EDGES DUE TO FLATTENING). THIS IS OF NO STRUCTURAL CONCERN

GRUMMAN

1590-031(T)

The AOTV vehicles experience aerodynamic heating effects at hypersonic velocities in the super-orbital speed regime. In addition to a significant convective input from the high energy boundary layer, a considerable radiative component may exist (on some AOTV configurations) due to the degree of ionization in the boundary layer. Historically, the thermal protection systems designed to protect a vehicle's primary structure from this heating have bonded a layer of ablative material or reuseable surface insulation to a continuous structural shell. These adhesive systems have had maximum use temperatures of approximately $500^{\circ}F$, but because of considerations for adequate margins of safety, particularly in manned vehicles, the bondline is restricted to $350^{\circ}F$. This has usually permitted the use of aluminum structure, which was attractive from an economic standpoint. It can be shown, however, that significant weight savings are possible in the vehicle primary structure if advanced materials are utilized.

The figure lists materials that appear to offer weight savings opportunities to AOTV. Composite materials have high strength to weight ratios. Metal matrix composites allow substantially higher structural temperatures, with a significant reduction in thermal protection system weight. Advanced aluminum alloys offer about 10% weight savings over conventional aluminum structures.

POTENTIALLY ATTRACTIVE ADVANCED STRUCTURAL MATERIALS

COMPOSITES

- GRAPHITE/RESIN
  - GR/EP
  - GR/BMI
  - GR/PI
- METAL MATRIX
  - BORON/AL
  - GR/AL
  - SIC/AL
  - SIC/TI

ADVANCED ALUMINUM ALLOYS

- POWDER METALLURGY AL–FE–CE, MO
- ALUMINUM LITHIUM

GRUMMAN

1590-032(T)

The figure displays some significant characteristics of graphite/resin composite materials. The major differences among these materials are their temperature limitations and their availability. Graphite/epoxy has the lowest maximum allowable temperature (350^{o}F) but is readily available with a significant amount of aerospace user experience. The highest maximum allowable temperature (600^{o}F) is for graphite/polyimide. This material has not had significant production experience.

COMPOSITE MATERIALS

GRAPHITE/RESIN ρ = 0.055 LB/IN.^{3}, E = 12 X 10^6 PSI, FTU = 75 KSI

GR/EP 350°F (DRY) $55/LB (WITH WASTAGE)
GR/BMI 450°F (DRY) $60/LB (WITH WASTAGE)
GR/PI 600°F (DRY) $65/LB (WITH WASTAGE)

- HIGHER TEMPS REDUCE TPS WEIGHT
- REQUIRE USE OF H/C FOR EFFICIENT SHELL STRUCTURES
- EXPERIENCE IN FUSELAGE TYPE STRUCTURES IS LIMITED FOR GR/PI
- LIMITED HEAT SINK CAPACITY BECAUSE OF LOW THERMAL DIFFUSIVITY
- LOW THERMAL STRESS EFFECTS

GRUMMAN

1590-033(1/2)

Some characteristics of four metal matrix materials are described on the figure:

- Boron fibers in an aluminum matrix
- Graphite fibers in an aluminum matrix
- Silicon fibers in an aluminum matrix
- Silicon fibers in a Titanium matrix

COMPOSITE MATERIALS (CONT.)

METAL MATRIX COMPOSITES

BORON/AL	ρ = 0.096 LB/IN.^{3}, E = 23 X 10^6 PSI, F_{TU} = 110 KSI 800°F, $150-200/LB
GR/AL	ρ = 0.083 LB/IN.^{3}, E = 16 X 10^6 PSI, F_{TU} = 65 KSI 800°F, $150/LB
SIC/AL	ρ = 0.104 LB/IN.^{3}, E = 22 X 10^6 PSI, F_{TU} = 80 KSI 800°F, $200/LB
SIC/TI	ρ = 0.134 LB/IN.^{3}, E = 28 X 10^6 PSI, F_{TU} = 87 KSI 1200°F, $500/LB

- STRUCTURES TECHNOLOGY IS IN DEVELOPMENT STAGE
- REQUIRES USE H/C FOR EFFICIENT STRUCTURES
- HIGH TEMP BONDING OR BRAZING FOR MAX TEMP
- EXTENSIVE FABRICATION STUDIES REQUIRED
- INTEGRATION WITH TPS REQUIRES EXTENSIVE STUDY

GRUMMAN

1590-033(2/2)

The figure displays some characteristics of advanced aluminum alloys. Some allow $500^{o}F$ temperatures. Others permit lighter aluminum structures at $350^{o}F$.

ADVANCED ALUMINUM ALLOYS

POWDER METALLURGY $\rho = 0.107$ LB/IN.^{3}, E = 13 X 10^6 PSI, F_{TU} = 75 KSI

AL-FE-CE (ALCOA) }
AL-FE-MO (P&W) } 500°F, $20/LB

- FABRICATION COST HIGHER THAN AL, LOWER THAN TI
- COMPETITIVE WITH TI AND GR/PI
- CONVENTIONAL DESIGN APPROACHES POSSIBLE
- PRODUCTION SCALE-UP REQUIRED
- MAY BE SUBJECT TO THERMAL STRESS PROBLEMS

<u>ALUMINUM-LITHIUM</u> $\rho = 0.090$ LB/IN.^{3}, E = 11.6 X 10^6 PSI, F_{TU} = 72 KSI

- NOT PRIMARILY A HIGH TEMP ALLOY, T_{MAX} = 350°F
- LOW DENSITY AND HIGH MODULUS MAY OFFER ADVANTAGES IN LOW TEMP AREAS

GRUMMAN

1590-034(T)

An analysis was conducted of typical aircraft fuselage structures to determine what proportion of the structural weight was designed to the four strength criteria shown in the left column of the figure. The right column displays the results of this analysis. The vast majority (60%) of weight was designed by ultimte tensile and compressive strength.

The different strengths-to-weight ratios of material (for the different failure modes) can be combined in the proportions shown on the right side of the figure to form a strength-to-weight ratio (with respect to some reference material) for a candidate material, at a given temperature, for typical aircraft fuselage type of structure. This procedure was followed for several materials, over a range of temperatures. The results are plotted on the next page.

STRENGTH CRITERIA FOR A/C FUSELAGE STRUCTURES

STRENGTH CRITERIA	% STRUCTURAL WEIGHT
TENSILE/COMP. STRENGTH	60%
CRIPPLING	10%
BUCKLING	20%
STIFFNESS	10%

GRUMMAN

1590-039(T)

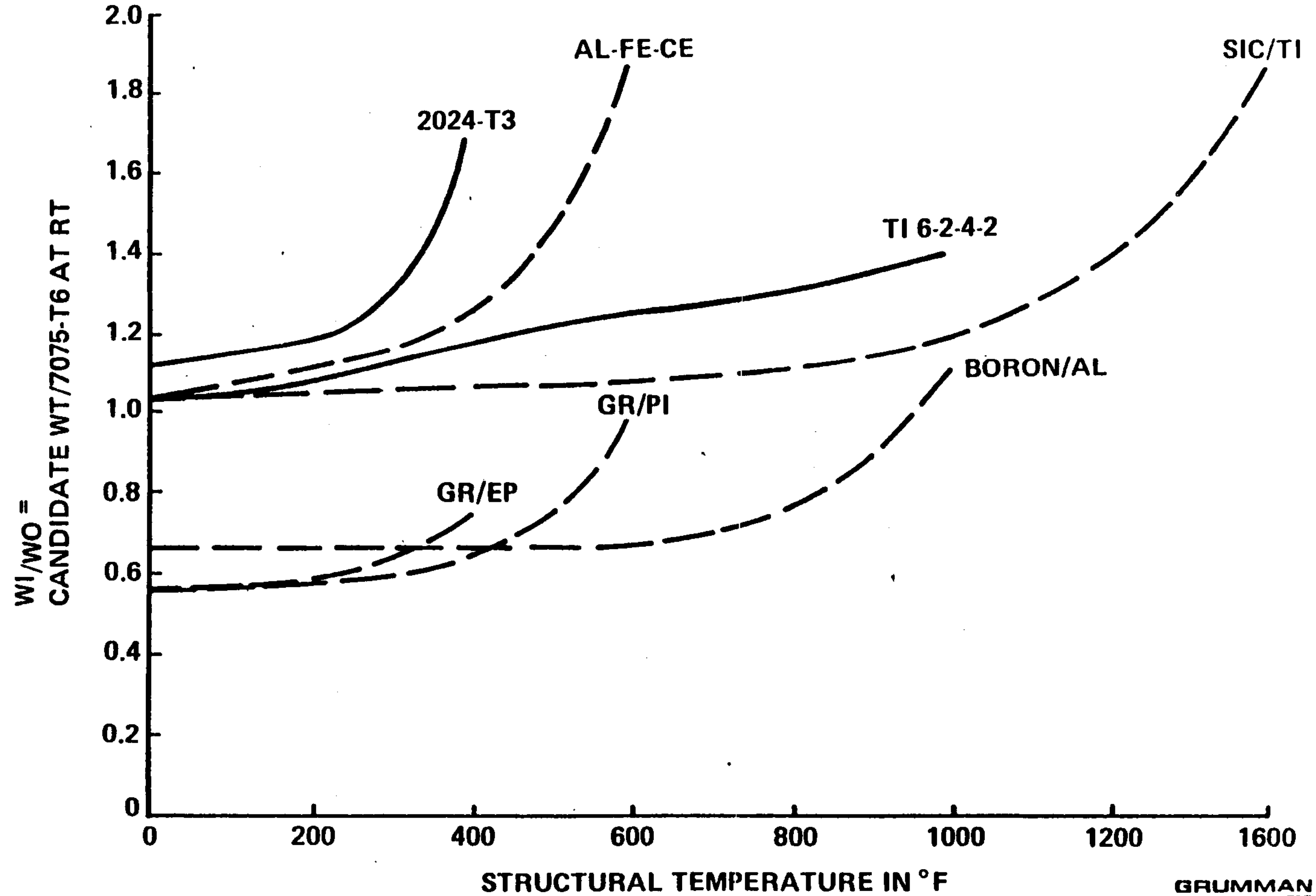
The figure shows the effectiveness of a variety of advanced materials over a range of temperatures, relative to the performance of a standard aircraft structural material (7075-T6) at room temperature. The abscissa contains a temperature range from $0^{o}F$ to $1600^{o}F$. The ordinate shows the ratio of the weight of an advanced material, at temperature, to the weight of 7075-T6 at room temperature, where both structures are designed to the strength criteria of the previous figure. For example, silicon fibers in a titanium matrix (SiC/Ti in Figure T39) have about the same strength through a temperature range of $0^{o}F$ to about $700^{o}F$, and this strength is approximately the same as 7075-T6 at room temperature ($Wi/Wo \approx 1.0$).

The figure shows that composite materials look attractive, particularly the metal-matrix-composites (MMC) such as boron/aluminum. Other MMC combinations, graphite/aluminum for example, are equally attractive but have not been plotted on the figure. These materials appear to have the potential for providing lightweight structure up to temperatures of at least $600^{o}F$. This increase in allowble structural temperature results in a decrease in thermal protection system weight. Furthermore, MMC's have the advantage of low coefficients of thermal expansion. This means that the increased structural temperatures will not result in an increase in thermal stress problems. Also, unlike resin-matrix systems, MMC's are not sensitive to outgassing in a hard vacuum at elevated temperatures, nor are they sensitive to ultra-violet radiation.

The MMC's are in a developmental state and additional technology development work is required. The basic material costs are high (\$150 to \$200 per lb). For a vehicle such as AOTV, this might not be an issue since the number of vehicles is small and improved performance has a large economic value.

Development of the most effective application of MMC materials to the primary structure of a typical AOTV vehicle should be undertaken. This would involve a design trade-off of structural temperature vs. total weight to determine the optimum TPS structure combination. Also, fabrication techniques applicable to MMC materials would have to be surveyed to identify viable candidate structural configurations. This would involve factors such as formability and joining techniques. In addition, a strength criteria will have to be studied since MMC's have limited ductility in certain types of loadings. Factors such as damage tolerance, fracture mechanics and fatigue criteria, and their effect upon design philosophy should be addressed if a realistic assessment is to be made for this application of MMC's. The end result will be an in-depth evaluation of primary vehicle structural weight savings potential based on MMC use. Also this study would contribute specific guidelines to aid in planned MMC material development for applications to high speed aerospace vehicles for future programs.

PROJECTED PERFORMANCE OF ADVANCED MATERIALS FOR FUSELAGE STRUCTURES



GRUMMAN

1590-035(T)

3.6 Technology Payoffs

The effect of subsystem weight reduction and other performance improvements on the AOTV propellant transport cost to LEO have been evaluated and summarized in Table A. Note that the recommended delivery mode of perigee kick + AKP provides cost sensitivities much different than the single stage delivery or manned round trip. This is due primarily to the much smaller perigee kick vehicle. The numerous potential subsystem weight reductions identified in Phase I of this study are still valid for this space based phase of the study (Phase II).

The mid L/D AOTV propellant transport cost sensitivities of Table A have been combined with the subsystem weight reduction possibilities to generate the propellant transport cost savings summarized in Table B. In this comparison, the perigee kick vehicle has been used for 10 GEO delivery missions per year and a single stage vehicle used for 2 manned missions per year.

Table A. Space Based Mid L/A AOTV Performance Sensitivities

	PERIGEE KICK	SINGLE STAGE	14K ROUND TRIP
$\frac{\partial(W_P + W_{AKP})}{\partial W_{DRY}}$ $(\frac{LB}{LB})$	0.73	2.8	2.9
$\frac{\partial W_P}{\partial I_{SP}}$ $(\frac{LB}{SEC})$	63.3		L/D = 0.75: 290; 1.5: 293
NUMBER OF FLIGHTS IN MODEL	100	100	20
$\frac{\partial \$ \text{ PROP TRANSPORT}}{\partial W_{DRY}}$ $(\frac{\$}{LB})$	$73,300	$282,000	$57,600
$\frac{\partial \$ \text{ PROP TRANSPORT}}{\partial I_{SP}}$ $(\frac{\$}{SEC})$	$6.3M		$5.8-5.86M

Table B. SPACE BASED MID L/D AOTV TECHNOLOGY PAYOFFS

(IN ORDER OF IMPORTANCE)

INCREASED I_{SP} (480 VS 443)	$451M
AVIONICS WEIGHT REDUCTION	$44-61M
EXTERNAL TPS DESIGN	$ 45M
ELE TRICAL POWER SUBSYSTEM WT REDUCTION	$18-33M
STRUCTURE WEIGHT REDUCTION	
IMPROVED ALLOWABLES + NEW MATERIALS	$ 9-26M
SPACE BASED VS GROUND BASED	$35M

* ASSUMES PROPELLANT TRANSPORT COST OF $1000/LB TO LEO 100 GEO-DELIVERY + 20 MANNED ROUND TRIP FLIGHTS

Additional technology advance benefits in the areas of improved aerodynamics and GN&C have been identified in the ground based Part I of this study and are still considred applicable here. The importance of some of the technology issues more nebulous to quantify for the ground based mode, is still applicable for the space based mode.

In addition, during Phase II, an AOTV-payload manifesting study was conducted to evaluate the relative advantages/disadvantages of alternate storable propellants. Results of the AOTV-payload manifesting study indicated that on a performance basis, 1) for a ground based only system, the storable propellant, N_2O_4/MMH, required fewer STS flights, 2) for a space based only or space based with ground based capability, the cryogenic propellant LO_2/LH_2 required significantly fewer STS flights. However, on the basis of a total cost estimate, including providing space basing capability, over a six year cost cycle, N_2O_4/MMH saved about $100M when compared with LO_2/LH_2. It is recommended that this area be examined in greater detail.

Considering all of the above, the recommended space based AOTV Technology Priority order is summarized in the Table.

SPACE BASED MID L/D AOTV TECHNOLOGY PRIORITIES

MISSION ENABLING TECHNOLOGY

AUTOMATION OF ROUTINE INSPECTION & MAINTENANCE

MISSION ENHANCING TECHNOLOGY

PRIORITY	ITEM
1	IMPROVED LIFE TIME OF STORABLE PROPELLANT ENGINE
2	AVIONICS WEIGHT REDUCTION + GN&C
3	EXTERNAL TPS DESIGN
4	AERODYNAMIC KNOWLEDGE
5	AEROTHERMODYNAMIC KNOWLEDGE
6	ELECTRICAL POWER SUBSYSTEM WEIGHT REDUCTION
7	STRUCTURE SUBSYSTEM WEIGHT REDUCTION

In October of 1984, visits were made to NASA LRC, JSC, and ARC to ascertain their perceptions of AOTV Technology Needs. The 1982 Aeroassist Working Group Technology Development Plan was used as a basis for discussing their current R&T Programs and Plans. As a result of this series of meetings, a list was prepared of those Technology Areas perceived to need supplemental emphasis/funding.

TECHNOLOGY AREAS NEEDING SUPPLEMENTAL FUNDING

	NEW INITIATIVE REQUIRED	AUGMENT ON-GOING PROGRAM
- DEVELOPMENT OF LESS CATALYTIC THERMAL PROTECTION MATERIALS COATINGS	X	
- TPS/STRUCTURAL DESIGN - EVALUATE NEW STRUCTURAL MATERIALS AND BOND SYSTEMS OPERATING AT HIGHER SOAK OUT TEMPERATURES	X	X
- DESIGN, FABRICATE AND TEST TRANSPIRATION COOLED NOSE WITH SEEDED COOLANT	X	
- AEROTHERMODYNAMIC METHODOLOGY - LEESIDE AND BASE AREA HEAT TRANSFER AND WAKE CLOSURE		X
- AERODYNAMICS - CONTROL FLAP EFFECTIVENESS		X
- AUTOMATION OF INSPECTION AND ROUTINE MAINTENANCE		
- PAYLOAD MANIFESTING ACROSS MISSION MODEL TO EVALUATE CRYO VS. STORABLE PROPELLANT TRANSPORT ADVANTAGES/DISADVANTAGES	X	
- GN&C - OPTIMIZATION OF A HYBRID FLIGHT CONTROL MECHANISM THAT BLENDS THE AERODYNAMIC AND REACTION CONTROL SUBSYSTEMS ON A LIFT MODULATED AOTV		X
- GN&C - DEVELOPMENT OF ADAPTIVE OPTIMAL STEERING LAWS		X
- AVIONICS - EVALUATION OF INERTIAL INSTRUMENT DEVELOPMENT AND PERFORMANCE, PROVIDE DIRECTION TO INSTRUMENT CONTRACTORS		X
- ELECTRICAL POWER SUBSYSTEM WEIGHT REDUCTION		X
- N_2O_4 - MMH ENGINE NEEDS NOT EVALUATED DURING THIS STUDY		
- MANUFACTURING METHODS FOR LOW COST DROP TANKS	X	
- STRUCTURAL HEALTH MONITORING	X	

3.7 Phase II Study Conclusions

The numerous major conclusions from the Space Based Mid L/D AOTV System Technology Analysis Study are summarized on the next three charts.

MAJOR PHASE II SPACE BASED MID L/D STUDY CONCLUSIONS

TECHNOLOGY ISSUES

PROPULSION SUBSYSTEM

- o ADVANCED OTV NOZZLE STRUCTURE IS ADEQUATE FOR ORBITER CARGO BAY ENVIRONMENT
- o OPTIMUM ENGINE NUMBER (LOX-H_2) FOR ADVANCED TECHNOLOGY BASE HEATING/ WAKE CLOSURE IS 3 TO 6 ENGINES
- o RECOMMENDATION FOR ADVANCED LOX-H_2 ENGINES
  - TOTAL THRUST 8-13K LBSF
  - MAN RATED CARGO VEHICLE - 4-3000 LBF

PAYLOAD MANIFESTING

- o APPEARS TO BE SMALL ADVANTAGE TO LARGE (VS. SMALL) SPACE BASED AOTV
- o SPACE BASED AOTV SHOULD BE CAPABLE OF OPERATING FROM GROUND BASED MODE
- o AT THIS TIME, PROPELLANT FOR A CARGO TRANSPORT AOTV - SPACE OR GROUND BASED, SHOULD BE EARTH STORABLE, N_2O_4-MMH
- o MANIFESTING DATA BASE USED TO GENERATE THIS CONCLUSION WARRANTS EXPANSION AND IN-DEPTH EVALUATION

MAJOR PHASE II SPACE BASED MID L/D STUDY CONCLUSIONS

SYSTEM ISSUES

- o GEO DELIVERY CAPABILITY OF STS TRANSPORTABLE MAXIMUM SIZE PERIGEE KICK + AKP IS FAR IN EXCESS OF CURRENT AOTV MISSION MODEL REQUIREMENTS
- o OFFLOADING A STRIPPED MANNED AOTV VEHICLE FOR GEO DELIVERY RESULTS IN A 9800 LB PROPELLANT PENALTY PER MISSION FOR 10 FLIGHTS/YEAR FOR 10 YEARS @ $1000/LB = $1B
- o CONFIGURATION PREFERENCE IS DEPENDENT UPON PROPELLANT TRANSPORT COST AND ACCEPTABILITY OF DROP TANKS
- o PERIGEE KICK VEHICLES CAN FLY NEAR ONE PASS OVERSHOOT BOUND TO REDUCE PEAK SURFACE TEMPERATURES

TECHNOLOGY ISSUES

AEROTHERMODYNAMICS

- o BOUNDARY LAYER TRANSITION TO TURBULENT FLOW EVALUATIONS PRODUCE CONTRADICTORY RESULTS - DELIVERY VEHICLE ALL LAMINAR, BUT MANNED ROUND TRIP VEHICLE TURBULENT BY ONE CRITERIA.
- o PEAK SURFACE TEMPERATURES OF MID L/D AOTV'S ARE SIGNIFICANTLY LOWER
  - NEAR OVERSHOOT BOUND ENTRY COMPARED TO LARGE PLANE CHANGE
  - TOTALLY NON-CATALYTIC SURFACE COATING COMPARED TO PARTIALLY NON-CATALYTIC OR FULLY CATALYTIC SURFACE
- o SUBSTANTIAL UNCERTAINTY EXISTS IN MAGNITUDE OF HYPERSONIC BASE HEAT TRANSFER AND HEAT TRANSFER TO PROTRUDING NOZZLES
  - CURRENT TECHNOLOGY SUGGESTS MINIMAL NOZZLE PROTRUSION INTO SEPARATED FLOW REGION
  - ADVANCED TECHNOLOGY MAY PROVIDE ENLARGED ALLOWABLE ZONE (CFD, GROUND TESTS, CALIBRATION OF METHODOLOGY)
  - FLAPS SHOULD BE MOVED ONTO BODY IF POSSIBLE TO AVOID TRAILING FLAP INDUCED SHOCK GENERATION

AERODYNAMICS

- o SPACE BASED AOTV'S THAT EXCEED LAUNCH VEHICLE ENVELOPE ARE NOT REQUIRED. CONFIGURATION TRENDS OF LOWER TOTAL SURFACE AREA (INDICATOR OF WEIGHT PENALTY) AND LOWER SURFACE TEMPERATURES (LIGHTER TPS) LEAD TO AMOSS/BICONIC TYPE CONFIGURATIONS